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Volume V



HIGH VOLTAGE DESIGN GUIDE: SPACECRAFT

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| 20. ABSTRACT (Continue on reverse side if necessary and identify by block number) This report supplies the technical background and design techniques needed by an engineer who is designing electrical insulation for high-voltage, high-power components, equipment, and systems on spacecraft. The data in this volume is in addition to the material in Volume 4 of the final report "High Voltage Design Guide: Aircraft" which is common to the electrical insulation design for both aircraft and spacecraft equipment. The common data includes testing, tracking, partial discharges and corona. Unique material involves the semi vacuum environment, spacecraft charging, and multipactor. | | |

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FOREWORD

Presented herein is the Boeing Aerospace Company's Final Report covering work accomplished on Contract F33615-79-C-2067 for the period of September 24, 1979 through January 5, 1983. This contract is being performed for the Aero Propulsion Laboratory, Air Force Wright Aeronautical Laboratories, Air Force Systems Command, Wright-Patterson AFB, Ohio. The program is under the technical direction of Daniel Schweickart, AFWAL/POOS-2.

Personnel participating in this work for the Boeing Aerospace Company were W. G. Dunbar, the technical leader, and S. W. Silverman, the program manager.

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1.0 PROGRAM OBJECTIVES

The objectives of this program are as follows:

- a. Perform high voltage tests on capacitors, cable assemblies and parts, and coils.
- b. Design, fabricate, and evaluate a high voltage standard test fixture to be used for measuring the void content in various high voltage insulation systems.
- c. Specify and procure a 150 kV, 400 Hz power supply for partial discharge measurements.
- d. Update the Tests and Specifications Criteria Documents completed in U. S. Air Force Contract F33615-77-C-2054 to include the findings from the test article evaluations.
- e. Develop a high voltage generator test procedure.
- f. Update of the Airborne High Voltage Design Guide completed on U. S. Air Force Contract F33615-76-C-2008.
- g. Develop a Spacecraft High Voltage Design Guide.

2.0 SCOPE

The major tasks reported in the development of this Spacecraft High Voltage Design Guide are:

- o The identification of the unique environment associated with the spacecraft and the relationship of the environment to the high-voltage high-power electrical/electronic systems.
- o The unique design considerations encountered in various insulation systems for the materials and electrical/electronic components.
- o The many tests that must be considered and the instrumentation and application of the test equipment to the components and system.
- o A list of problems and solutions to these problems for high-voltage spacecraft equipment.

3.0 BACKGROUND

In a previous contract F33615-76-C-2008 "High Voltage Design Guide for Aircraft", high voltage/high power technology was evaluated for methods of utilizing high voltage power equipment in aircraft. That contract resulted in a design guide document which provided technical information in an area where such information did not exist. A subsequent contract F33615-77-C-2054 "High Voltage/High Power Specifications and Tests" resulted in the preparation of specifications for eight items used in high voltage power supply systems. The eight were selected from an extensive list of items for which no specifications existed. These were prepared for use as procurement specifications for advanced technology hardware. The present contract F33615-79-C-2067 "High Voltage Testing", deals with the conduct of tests to verify the specifications in contract 2054. This phase of the "High Voltage Testing" contract is to develop a "Spacecraft High Voltage Design Guide" to extend the work for the "High Voltage Design Guide for Aircraft" to space applications.

Planning charts for future spacecraft programs show that the projected electrical loads are for multiple kilowatts, up to megawatts as the advanced payloads are being developed. With the larger systems the spacecraft will also be larger, necessitating longer cable runs so that the electrical transmission and distribution will become similar to compact electric utility systems in space. To distribute power over longer distances, it will be necessary to increase the voltage and decrease the current, and thereby the losses. Increased voltage leads to high voltage problems and the development of a distribution, protection, and control system for spacecraft concepts which affect the electrical power system. For one thing, spacecraft structures are being developed using carbon fiber epoxy, which requires a reevaluation of methods of distribution and protection because the structure is essentially non-conductive. Also, at the high power level and high distribution voltage, the generated electric and magnetic fields are intense enough to interact with the space environment, causing breakdowns or degradation. The technology of high-power and high-voltage in spacecraft thus surfaces many technology problems to be solved.

Increased activity in high voltage and high power for spacecraft pushes the technology of high voltage even further than for aircraft because of the more severe effects from long term operation in a space environment, often without any manned operation or maintenance. Spacecraft problems can be more severe because of outgassing,

radiation, space particulates, solar wind and solar flares, and plasma interactions. Spacecraft also include devices which cannot be protected by shielding (e.g., solar arrays), and weight becomes many times more restrictive than for aircraft. To reduce weight, many practices used for aircraft or terrestrial equipment cannot be used for spacecraft. The lifetimes for spacecraft components are more difficult to satisfy because the equipment usually operates unattended and cannot be inspected. For an aircraft, parts can be checked, inspected, repaired, etc. to derive long periods of operation. Such luxuries are usually not possible once a spacecraft is put into orbit, unless the spacecraft is retrieved or redundancy provided. Manned spacecraft can consider maintenance and repair in orbit. Some methods of retrieving certain unmanned spacecraft are now being considered.

We are also beginning to see spacecraft which are significantly larger than any aircraft. Wire runs can be hundreds of feet long and power levels are beginning to rival and sometimes even exceed power levels for aircraft electronics. Of course, not all the technology developed for high voltage/high power systems for aircraft is useless for spacecraft. A large percentage of the material in the "High Voltage Design Guide for Aircraft" is applicable to spacecraft power systems. But, because of the unique differences in the environment and some components (e.g., solar panels), new technology must still be developed for a spacecraft high voltage design guide.

Spacecraft containing high voltage/high power components and distribution systems have multiple problems which require special design techniques not normally used in ground equipment or in aircraft electrical systems. This results from the unique environment, the long flight duration, and the lack of maintenance (of unmanned satellites) of the spacecraft during its mission. In low power spacecraft applications it was found that the application of electrical insulation technology to component design resulted in improved materials which were capable of withstanding much higher electrical stresses than were used in ground support and airplane systems because the designers realized the significance of outgassing, thermal stresses, and high voltage field stresses. Even with these advances in materials and reduction of field stress points there are still the problems of design in spacecraft containing medium power and medium voltage electrical systems and components. These problems will be amplified as the voltage and power levels are increased. Therefore, it is necessary that designers have a better understanding of high voltage design principles before attempting a high power application design using high voltage. Once the correct

design principles are established and the hardware developed, the design must then involve many technical disciplines - mechanical design, electronics design, materials applications, structures, packaging, test technology, and the electrical system design, all in conjunction with the high voltage specialist. The team must consider the total environment in which the system must operate. For example, wire insulation is expected to operate in space vacuum after being subjected to months of shelf storage on Earth. The wiring may have been flexed, temperature cycled, vibrated, manhandled, and even exposed to hostile fumes. These mechanical and chemical stresses, in addition to the electrical stress, can lead to deterioration and eventual breakdown of the insulation.

4.0 ENVIRONMENT

One aspect of electronic design that is easily overlooked is the provision for partial discharges and voltage breakdown at low gas pressures. Throughout the space program there have been occasions when loss of equipment due to partial discharges and voltage breakdown has resulted in partial mission success. This summarizes some of the important causes of high voltage problems in space flight so that designers of high-voltage spacecraft equipment will not repeat any of the costly experiences of the past.

4.1 Pressure. Most satellites operate in space where the low pressure (less than 10^{-4} N/m²) makes the theoretical dielectric strength of the volume of gas greater than 5×10^5 volts/cm, a value 16 times the dielectric strength of dry air at sea level. This is because there are few carriers and the mean free path exceeds the gap length between closely spaced electrodes. But the spacecraft and its modules and circuits slowly outgas during boost to orbit and after the spacecraft is inserted into orbit. For instance, as the spacecraft is boosted into orbit, the external pressure decreases rapidly from Earth sea-level pressure (1.013×10^5 N/m²) to less than 1×10^{-5} N/m². This pressure change takes only a few minutes, but the pressure next to the outer surface and inside the spacecraft will remain at higher pressure throughout the life of the spacecraft, due to the outgassing of various materials.

4.1.1 Internal Gas Pressure. During boost, gas escapes rapidly from the spacecraft interior for the first thirty kilometers into space while continuum gas flows. This is due to the slow outgassing through small orifices, tubes, and cracks, of gases entrapped in electrical and thermal insulations and structural materials.

If a spacecraft has no outgassing products, the flow of gas can be calculated by the Clausing equation¹ which is used for estimating the flow of gas from chamber to chamber in a multiple chamber vacuum system.

$$C = 3.638 A_k (T/m)^{1/2}$$

Where:

C = flow conductance for the orifices in cc per second

A_k = area of the k orifices in square centimeters

T = internal temperature in °K

m = mass of the gas molecule in grams

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Scialdone² calculated and measured spacecraft compartment and equipment outgassing rates. He shows that the depressurization time constant, t , which is the time for the pressure to decrease to 36% of its initial pressure, is V/C , where V is the volume in cubic centimeters and the conductance C is:

$$C = K \bar{V} A_k$$

$$\bar{V} = (8 kT/\pi m)^{1/2}$$

Where:

\bar{V} = molecular flow speed in cm per second

k = Boltzman constant.

Gases such as air and nitrogen have a time constant of about 0.4 second when bled through a 1-cm² opening in a 10-cm radius steel sphere³. NASA experience² has shown that a 0.1s time constant insures adequate outgassing around high voltage circuits.

These equations work well with known outgassing port sizes, spacecraft volumes, and non-gassing parts. However, most spacecraft have thermally insulating coatings, fibrous insulation, electrically insulated parts, semi-shielded and electromagnetically-shielded boxes, and boxes within modules. In addition, compressed gases for orbit keeping are stored on the spacecraft and released. With these many gas sources, it is often better to qualify the design by testing the completely assembled spacecraft in a thermal vacuum chamber than to measure the real internal and external spacecraft pressures and the outgassing rate.

4.1.2 Outgassing Through Insulation. An experiment was designed to show the relationship between the outgassing rate of an electronic component under a thermally insulating blanket versus that in the outside atmosphere. In this laboratory experiment, a thermal blanket having 100 layers of super-insulation was placed across the center of a vacuum chamber. Gas flowed through the interstitial spaces in the insulation. During the first fifteen minutes of pump-down, the pressure in the chamber dropped from sea level to 10 N/m², with the gauge on the thermally insulated side of the chamber following the pump pressure within 5% as shown in Figure 1. As

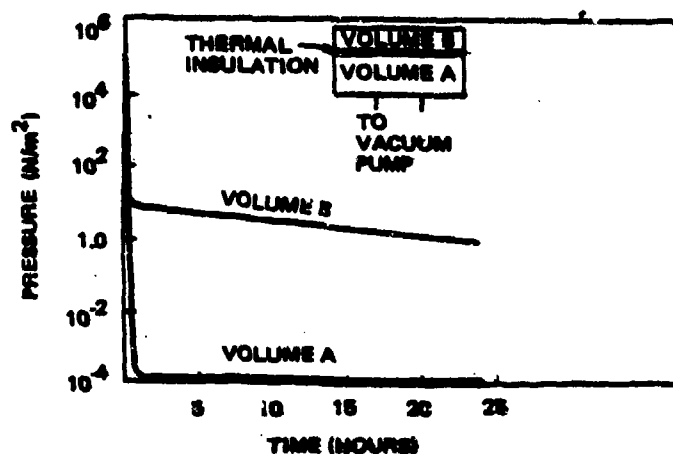


Figure 1: Effects of Thermal Insulation on Outgassing Rate

the pump pressure dropped further, the pressure at the insulated side of the chamber decreased very slowly.

In the Skylab Apollo Telescope Mount, the outgassing area was measured to be about one square centimeter per liter volume, the value recommended for adequate spacecraft outgassing when high voltage experiments or equipment are on board. The resulting pressures reported^{4,5}, are summarized in Figure 2.

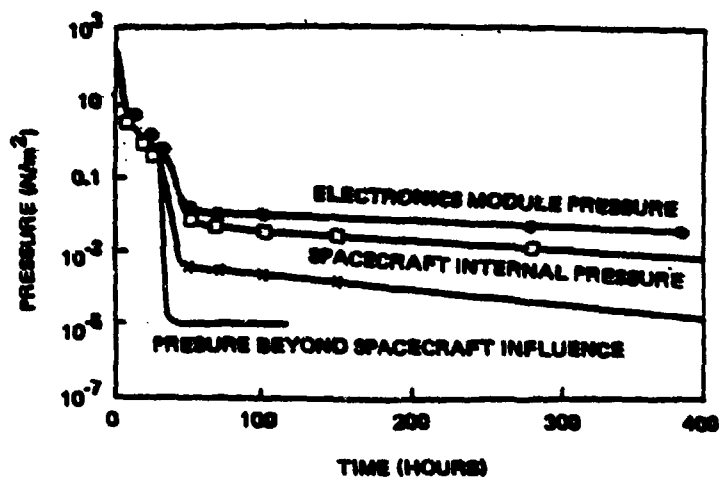


Figure 2: Gas Pressure Inside the Apollo Telescope Mount

Cuddihy and Moacanin⁶, in measuring the outgassing rates of polyurethane foam, used for electrical/electronic insulation, found that the calculated value based on the reported permeation constant of the measured value agreed within a factor of 2. The diffusion coefficient (D) for a foam is calculated with the equation:

$$D = P_e \left(\frac{RT}{m} \right) \frac{P_0}{P} \left[\frac{1}{(1-P/P_0)^{1/3}} + 1 + (1 + P/P_0)^{1/3} \right]$$

Where:

D is the diffusion coefficient in cm³/sec

P is the foam density in g/cc

P_e is the permeation constant in mm/s/cm²torr

P₀ is the density of the bulk polymer in g/cc

R is the gas constant.

T is internal temperature in °K

m is mass of the gas molecules in grams

They also found that for sufficiently long outgassing time, the weight loss of the gas, when plotted as a function of the thickness of the solid (A), eventually becomes linear with the slope of (π/2A)² D.

The superinsulation, Apollo Telescope Mount, and polyurethane form test experiments show that the spacecraft internal pressure is significantly greater than the external pressure for several days after orbit insertion. Furthermore, outgassing products within a high-voltage module may keep the pressures much too high for safe, reliable operation of high voltage circuits, making it advisable to delay their TURN ON. Likewise, the outgassing products of the spacecraft and reaction control propellants increase the pressure in the vicinity of the spacecraft. For very high-power high-voltage equipment it may become necessary to package in pressurized gas or oil to prevent corona and/or voltage breakdown due to slow outgassing and pressure.

4.2 Particulates. Many future space missions will require large spacecraft to support the onboard loads⁷. Some of these spacecraft will require kilowatts to megawatts of electrical power. Even though the amount of cosmic dust flux is very small in low Earth orbit to geosynchronous orbits, the effects of cosmic dust on large spacecraft are significant enough to produce problems with high-voltage systems. Various predictions on the effects of space environment on large spacecraft have been

based on experimental experience gathered from small satellites^{8,9,10}. However, little attention was paid to the effect of particulate debris on the power systems of these spacecraft, probably because there is no evidence that particulate space debris has caused problems for low-voltage small satellites. A brief preliminary analysis of the expected problems that the power system of a large spacecraft might experience as a result of contamination by particulate debris has been made¹¹.

4.2.1 Sources Of Particulate Debris. Spacecraft are impacted by particulate debris from three sources:

- (1) Earth environment. Debris from Earth environment are mostly dust and sand particulates and particulates from rocket exhaust¹². This contamination is easily minimized by implementing existing techniques for high-standards and control, therefore Earth environmental effects are not considered in this analysis.
- (2) The spacecraft itself. In addition to the degradation caused by the impact of debris, outgassing of some of the materials from the spacecraft will generate more debris. Possible sources for such debris are the glass cover for the solar cells, the solar panel, adhesives, the slipring assembly of the power system, electric insulating materials, thermal blankets, propulsion engines, and the main structure. In addition, spacecraft ion thrusters emit ions that contaminate the space plasma. D. Heier's¹³ experimental investigations on a slipring assembly of satellite indicated that the lowest wear rate of the brushes was 0.000594 in. per year. This corresponds to an insignificant amount of carbon debris, and its effects are ignored. Quantitative information on the other contaminants (with the exception of the solar panel) is not available, therefore the results presented here correspond to the lower limit of the expected amount of debris.
- (3) Space environment. It should be remembered that outer space is not an absolute vacuum. There are residues of gases that form the space plasma, and also particulate debris that are of extraterrestrial origin. The term "cosmic dust" refers to particulate debris with masses $\leq 1.0g$; i.e., micrometeoroids and small meteoroids. Extensive studies have been conducted on cosmic dust (e.g. References 14-17). Following is a brief listing of parameters that are pertinent:

- o Average value of cosmic dust speed relative to the Sun is 18.5 km s^{-1} .
- o Average value of micrometeoroid cumulative flux at geostationary orbit with mass $\geq 10^{-12} \text{ g}$ is $182 \text{ particles m}^{-2} \text{ day}^{-1} (2\pi \text{ sr})^{-1}$, where $(2\pi \text{ sr})^{-1}$ is the flux of a particulate that sticks to a plane surface.
- o The NASA model that describes the average annual cumulative micrometeoroid flux is given by the logarithmic formulas:

$\text{Log } N_T = -14.37 - 1.213 \log m$ for the range $10^{-6} \leq m \leq 1.0 \text{ g}$.

and

$\text{Log } N_T = -14.339 - 1.584 \log m - 0.063 (\log m)^2$ for the range $10^{-12} \leq m \leq 10^{-6} \text{ g}$
 where N_T is the number of particles $\text{m}^{-2} \text{ s}^{-1}$ of mass m or greater, and m is in grams.

It should be noted that we used average values for the cumulative flux and speed of cosmic dust because of the wide range of reported values.

4.2.2 Why Large Spacecraft Are More Susceptible To Particulate Debris Than Smaller Spacecraft. The design of a spacecraft, particularly of its electrical power system, and the purpose of its mission play an important role in its susceptibility to debris contamination. Large spacecraft with large electrical-power systems in the order of hundreds of kW are more subject to problems caused by particle debris than are smaller spacecraft because of the following characteristics, which are inherent in the design and the function of large spacecraft:

- o The high voltages and high currents of unshielded interconnectors, cables, and buslines will generate strong electric and magnetic fields that will strongly influence particulate debris-spacecraft electromagnetic interaction.
- o The large areas of dielectric materials in the high power solar array structure will produce strong electric fields because of surface charging and differential charging. This will tend to collect particulate debris.
- o The use of low-density materials in the spacecraft structure will yield some outgassing and fragmentation products.

o The requirement for long life, e.g. 10 to 20 years, will lead to cumulative effects of particulate debris impacts. These characteristics are barely evident in the design of small spacecraft, therefore their effects are not significant. The importance of the role played by these characteristics depends, however, on the amount of particulate debris. In the following section, it is shown how much debris may be expected and what the subsequent effects will be.

4.2.3 Overall Expected Debris and Possible Problems. The previous information on debris sources and on the above equations leads to an estimate of a total mass of $8.54 \times 10^{-7} \text{ g m}^{-2} \text{ day}^{-1} (2\pi \text{sr})^{-1}$ of particulate debris that is due to cosmic dust where the unknown amounts due to outgassing and ion thrusters are ignored. This very small amount may mislead designers who might assume that its effect on a spacecraft is insignificant. The following analyses show, however, that resultant problems are not only dependent on spacecraft configuration and on the total incident mass flux of particulate debris, but also are highly dependent on the velocities of the striking particulates. The speed, V_r , of a cosmic dust particle relative to a spacecraft in geosynchronous (GEO) orbit is given by

$$|\bar{V}_r| = (|\bar{V}_{sp}|^2 + |\bar{V}_D|^2 - 2|\bar{V}_{sp}||\bar{V}_D|\cos\theta)^{1/2}$$

\bar{V}_{sp} is the spacecraft velocity vector relative to the Sun; for a spacecraft at GEO, $\bar{V}_{sp} \approx 30 \text{ km s}^{-1}$, assuming a circular orbit around the Sun,

\bar{V}_D is the dust particulate velocity vector relative to the sun; it is assumed that \bar{V}_D equals the average value, i.e., $|\bar{V}_D| = 18.5 \text{ km s}^{-1}$, and

θ is the angle between the dust particulate velocity vector and spacecraft velocity vector.

This equation defines the range of speeds of a cosmic dust particle relative to the spacecraft; i.e., $11.5 < |\bar{V}_r| < 48.5 \text{ km s}^{-1}$. Knowing this velocity range, we may classify the overall impacting debris into three groups, depending on their energies. These groups and the overall problems expected with each are discussed in the following paragraphs.

4.2.3.1 High Energy Debris. Debris such as cosmic dust that strike the spacecraft at a very high velocity (e.g., $\geq 10 \text{ km s}^{-1}$) have large kinetic energies; therefore, after they impact the spacecraft, a shock wave propagates from the collision interface,

compressing and heating the affected areas. The particulate-spacecraft interface is melted, and a fraction of the material is evaporated and even ionized. This results in cratering and fragmentation in the surrounding region. Estimates of debris ejection, plasma production, and their effects are presented in the following:

Debris Ejection and Their Effects. The experimental work of Gault and Heitowit¹⁷ showed that an aluminum projectile impacting upon basalt surfaces with a velocity of 6.25 km s^{-1} induced ejecta of mass that were approximately 370 times the mass of the projectile. Dohnanyi¹⁸ and Marcus¹⁹ have indicated, however, that a large range of experimental results on ejecta generated from impacts on rocks are approximated by $M_e = 5 V^2 M_p k^2$, where M_e and M_p are the masses of ejecta (kilograms) and projectile, respectively; $V \text{ km s}^{-1}$ is the velocity of impacting particles relative to the targets; and k is a normalization constant $1.0 \text{ km}^{-1} \text{ s}$. Thus, for a velocity of 6.25 km s^{-1} , this equation would yield almost half the amount suggested by Gault and Heitowit, which further suggests that this equation presents a conservative estimate of ejecta generated by cosmic dust impacting the spacecraft. Using the previous information on cosmic dust speed and flux in the Dohnanyi and Marcus formula leads to an estimate of expected ejected mass from the solar panel. The ejected mass lies in the range $5.65 \times 10^{-4} \leq M_e \leq 10^{-2} \text{ g m}^{-2} \text{ day}^{-1} (2\pi \text{ sr})^{-1}$. This mass removal will cause a significant perturbation to the spacecraft orbit over the long period of operation and should be considered in the orbital analyses of the spacecraft. The produced ejecta are subjected to the electric and the magnetic fields that surround the spacecraft. Hence, debris, whether from space or from the spacecraft, will be swept (and collected at selected regions) by the fields of unshielded high-voltage/high-current conductors. Once the debris particles have entered the field of the conductor (Figure 3), they will become charged and polarized to form a seta (hairlike) growth on the conductors. Some of these charged particles will form bridges or "strings of pearls" between high-voltage and low-voltage or ground planes (Figure 4). The field stress at the ends of the seta is very high, so any plasma in that region can and will initiate an arc between the conductor and the ground plane or another conductor. This causes power losses and changes in physical characteristics. Other problems resulting from particulate debris contamination are:

1. Short circuiting of electrical elements.
2. Arcing and surface damages over the solar array and the main structure.

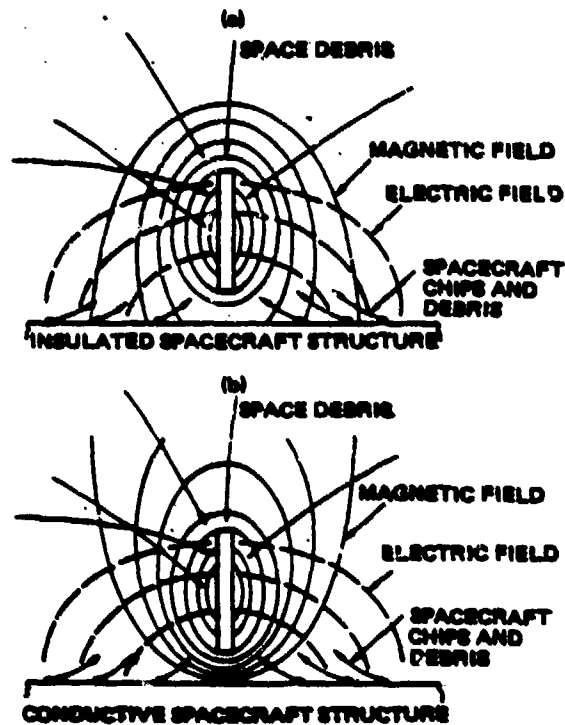


Figure 3: The Configuration of Electric and Magnetic Fields Will Influence the Trajectories and Hence the Collection of Debris. Fields Due To High Voltage/High Current Unshielded Cable Placed (a) Over an Insulated Surface, (b) Over a Conductive Surface.

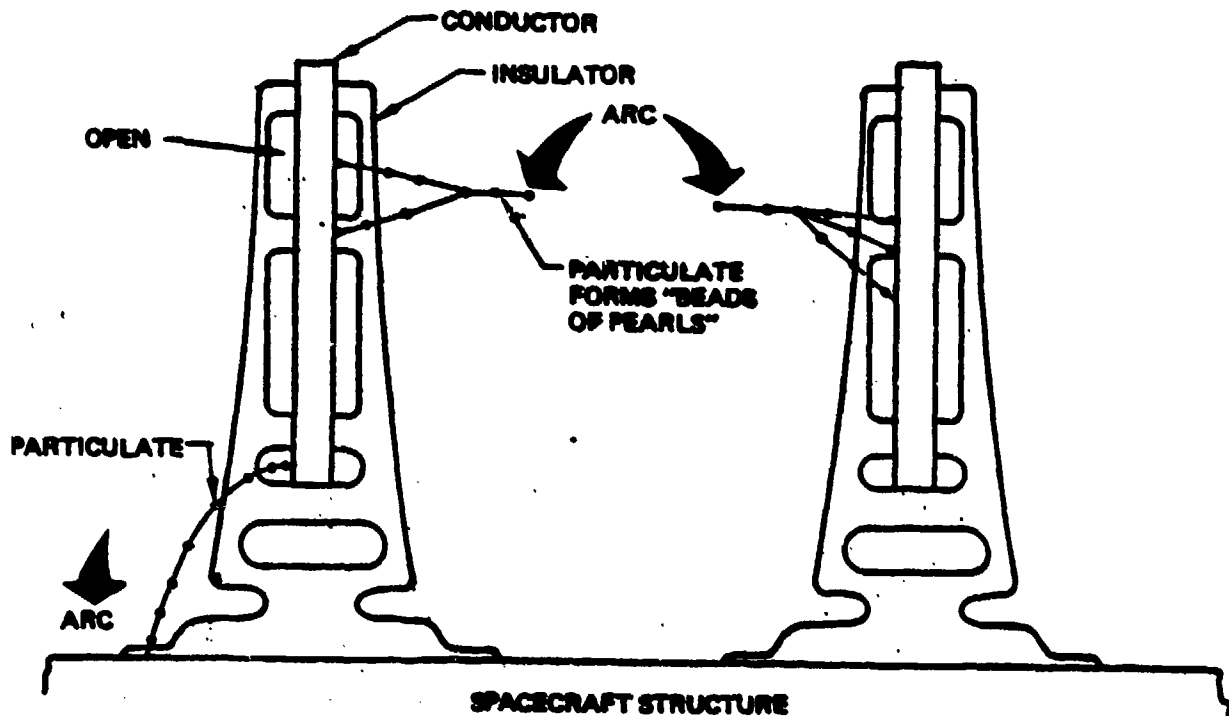


Figure 4: Arcs Caused by Particulate Bridge Between Conductors

3. Power leakage through the plasma due to the pinhole effect at the surface of the solar array.
4. Arcing on the main bus lines to communication antennas, which causes current irregularity and produces intolerable noise in addition to physical damage.
5. Degradation of paints and surface finishing.
6. Excessive wear and binding of movable parts, because of contamination of lubricants, seals, and sliding surfaces.

4.2.3.2 Low Energy Debris. Debris that are striking the spacecraft at lower velocities, e.g., 10 km s^{-1} , such as slowed-down micrometeoroid and some ejected fragments, will generate problems similar to those listed in 1 through 6, but at lower rates and of less intensity.

4.2.3.3 Settling Debris. Particulate debris that are settled on or that are suspended in the neighborhood of the spacecraft will be charged up such that at equilibrium conditions the potential at their surfaces, will equal that of the closest surface of the spacecraft. This leads to high potential at the particulate debris surface, especially if it is in the shadow or near a dielectric. Charges will build up on particulate surfaces until they discharge or until they break down because of the electrostatic pressure. This phenomenon of breakdown may be described by the semiempirical relation¹⁷:

$$F = 8.85 \times 10^{-7} \phi^2 / r^2$$

where

F is electrostatic stress on a tensile of a particulate in dyne/centimeters², ϕ is the particulate surface potential in volts, and r is the particulate radius in centimeters.

If F exceeds the particulate's tensile strength, the particulate will break apart, yielding smaller particulates. Thus, more small particulates are generated that eventually, over a long period, will cause optical shielding around and over the solar panels. Some particulates will tend to migrate to specific localized regions on the spacecraft because of the electromagnetic interaction, whereby setas are formed which eventually lead to shorting and/or arcing. Quantitative estimates of the produced debris are unavailable. Experimental investigation is required to determine the amount of particulates produced as a result of the breakdown of different

particulate materials (i.e., different composition, shape, size, etc.) that are subjected to a wide range of surface charging.

4.2.4 Controlling Incident Particulate Debris. In order to control damages induced by particulate debris, stronger but lighter materials must be developed to withstand the impacts of energetic particulates. Although many materials have been developed recently which satisfy this criteria, there are still limited successes, e.g., solar panels are still susceptible to mechanical damage by debris. This indicates the limitation of this approach and suggests the need for developing a complementary approach so that a satisfactory solution can be achieved. A method is to slow down, deflect, and/or to prevent particulate debris from impinging on the spacecraft. This may be achieved by utilizing the electromagnetic fields of the electric currents of the spacecraft. The electric charge of the particulate debris (and its magnetic moment if it possesses one) interacts with the electromagnetic fields and thus it is possible to control its motion relative to the spacecraft. This approach is similar to some approaches that were proposed for plasma shielding. Therefore, it is appropriate to present a brief review of this concept.

First, as an illustration Figure 5 shows the influence of the magnetic field of a conductor carrying a current in a moving plasma. The inertia forces of the accelerated plasma or a gravitation field induce a charge separation mode at the plasma boundary.⁽²⁰⁾ This plasma is confined in the magnetic field around the conductor. Parker⁽²¹⁾ and Oran had implemented this concept in their analyses for a solar power satellite. They considered a configuration where two cylindrical bus lines carry the current from a 2 x 20 Km solar array as shown in Figure 6. Their analyses indicated that electrons up to several Mev in energy are prevented from reaching a large fraction of the array surface. In the vicinity of the 100 kiloamp currents near the lower end of the array, the width of the protected portion of the array is 270 meters within which the lifetime of the solar cells would be prolonged by a possible factor of 5. Miller et. al ⁽²²⁾ had conducted similar analyses on a section of a solar array, but considered more complicated configurations for the current sources and the bus lines, see Figure 7. The magnetic field due to this current distribution was determined over the array. Their analyses had indicated that for normally incident plasma, most of the panel is not screened from electrons with energy greater than 1 Kev except along the crossed mid lines. This indicates that this configuration may

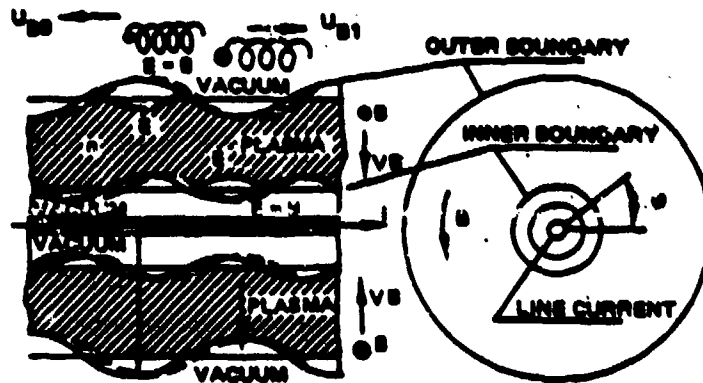


Figure 5: Cylindrical Shell of Plasma Confined in the Magnetic Field from an Axial Line Current

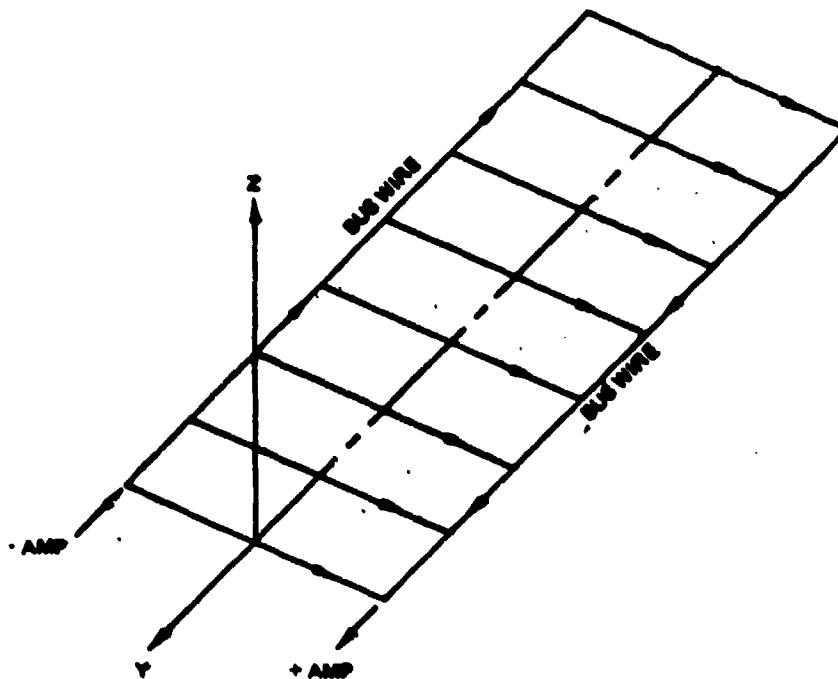


Figure 6: Current Distribution in Solar Array Panel Analytical Model

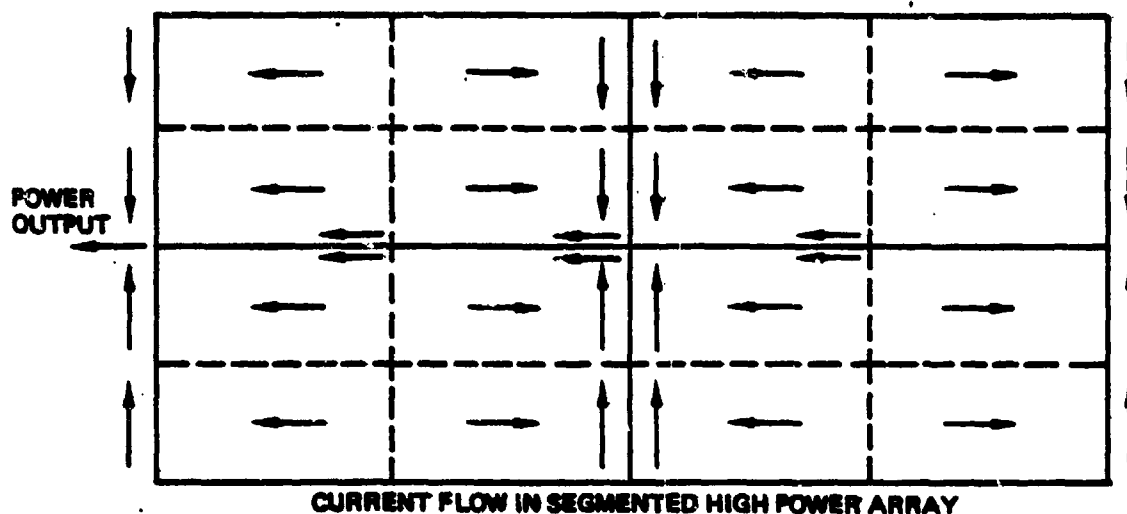


Figure 7: SPS Current Flow

promote differential charging and electron avalanche. For protons the cutoff energies are $1/1836$ of those for electrons. Hence, most of the panel is not protected against protons with energy greater than 1 eV. Thus, the analyses of Parker and Oran, and of Miller et. al. indicate the complexities of the shielding problem. An unlimited number of configurations and designs for the electric conductors, cables, and connectors may be implemented. Each has its shielding effects with varied advantages and disadvantages.

4.2.5 Controlling the Spacecraft Potential. As previously mentioned, energetic debris produce plasma bursts upon impacting the spacecraft. These plasma, the one produced by the ion thrusters, solar wind, the ambient plasma, and the photoelectric effect are continuously changing the electrostatic potential of the spacecraft through uncontrolled charging and discharging mechanisms. Obviously, to maximize the performance of the spacecraft, one has to limit the potential on its surfaces to acceptable values. Several interesting techniques have been developed for controlling the potential of small spacecraft. These techniques are based on the emission of charged particles using an active or inactive potential control system. As a demonstration for active control systems⁽²³⁻²⁶⁾, two techniques have been used; the hot wire filament electron emitter for the ATS-5 and the plasma bridge neutralizer for the ATS-6. Both systems share the concept of releasing electrons from the charged spacecraft. The plasma neutralizer of the ATS-6 had, however, another advantage;

ions from the discharged plasma in the neutralizer are attached to the neighboring insulating surfaces, therefore discharging them. For inactive systems, Grard, et al⁽²⁷⁾ and Grard⁽²⁸⁾ had investigated the possibility of using an electron field emitter. Its operation is based on charge dissipation in the ambient environment which primarily depends on the potential difference between the spacecraft and the environment. This potential difference forces electrons to leave sharp pointed filaments at the end of a conducting boom attached to the spacecraft. No power source is required to activate this mechanism, but the length of the boom has to be several times the dimensions of the spacecraft in order to minimize the influence of the induced charge (in the spacecraft) on electrons emission. In the above mentioned techniques, the interaction between the released electrons and the ambient environment is not fully understood. This adds another obstacle in addition to that for scaling in attempting to implement these techniques for large spacecraft. Out of the available analyses on large spacecraft, very few tackled the concepts for the reduction of plasma impacts. These are the previously mentioned analyses of Parker and Oran and of Miller et al, which are based on shielding rather than bleeding or neutralizing the parasitic charges. Excessive charges flow through this conducting system to designated sites. By properly coupling these sites, one may utilize the excessive charges in performing a specific task in addition to the control of the static potentials on the spacecraft surface. The selection of either type of conducting system for a design recommendation will be based on detailed trade studies which include performance, weight, cost, degradation, and power output at end of design life.

5.0 DESIGN CONSIDERATIONS

Each spacecraft high-voltage design is unique depending upon the topical design considerations discussed below. There are many minor areas of concern that must also be considered in addition to these major topics which fall directly or indirectly into one of the discussed subparagraphs.

5.1 Power Demand. Some future spacecraft systems for the post 1990 era will be designed to supply high power at high voltage to meet the ever-increasing on-board load demand²⁹. Key configurations for these large high-voltage multikilowatt systems must consider the power source, distribution subsystem, and each load and its relation to the load family and mission. Some loads are always totally independent of other loads, while others are used as ancillary equipment to support the mission. Some planned spacecraft systems have need for the assembly of large structures in space which have large electrical power systems to supply the electrical/electronic equipment loads. These electronic systems will be used for communications, radar, special electronics, experimental equipment, and future energy sources. Some of the systems (in the 1985-1990 period) will have power levels to 50kWe. The long term programs, 1990 to post-2000, will have much higher demands, possibly in the order of multi-megawatts.

Near and far-term missions are plotted with respect to mission power demands of a few manned and unmanned satellites in Figure 8. Near and far-term mission power demands will require that either the power systems have very large currents at low voltage or the voltage must be increased to keep the current levels and the losses down⁷. To meet the multimegawatt goals, both high current (over 1000 amperes) and high voltage must be considered in the same power system. A plot of the current and voltage relationship to power and advanced technology distribution equipment is shown in Figure 9. Three voltage regimes are shown in Figure 9: voltages to 200 volts, between 200 volts and 2000 volts, and over 2000 volts. There are some specifications and standards for the lower voltages between 0 and 200 volts, few for the transition voltage (T) regime (200 to 2000 volts), and very few for the high-voltage (HV) over 2000 volts. Ground return via cables and connectors on spacecraft composite structures will all be influenced by the voltage, current, and power level of the spacecraft. Vehicle size will affect the cable lengths, thus, the voltage drop and voltage variance across such items as solar panels. Secondary effects will include transients, traveling

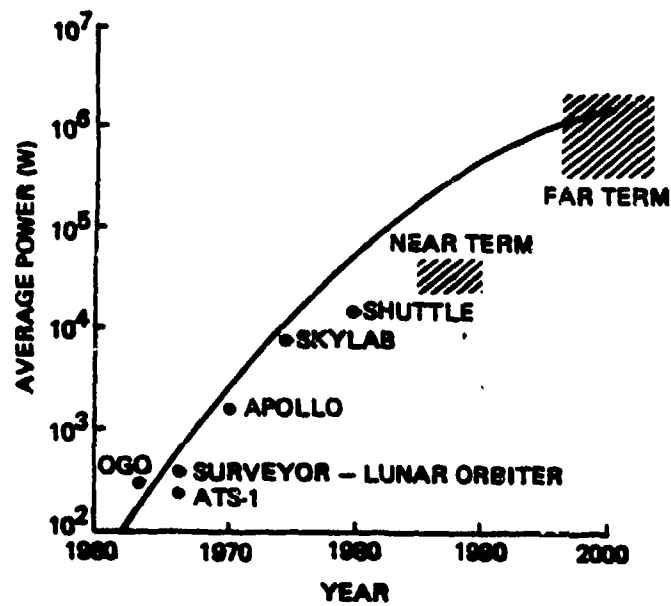


Figure 8: Mission Power Requirements

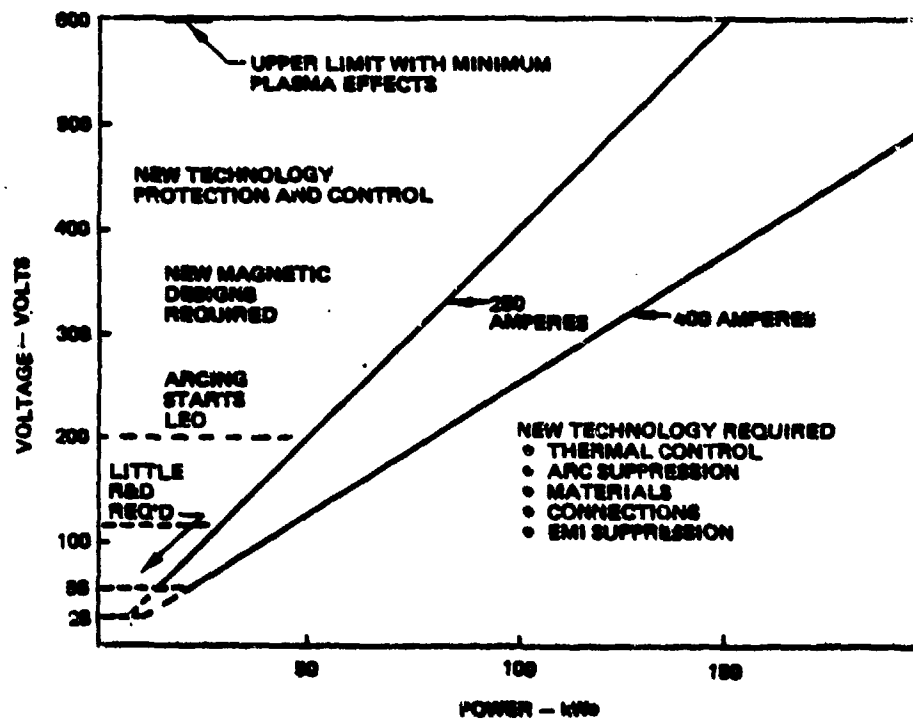


Figure 9: Power Distribution - Wire and Connectors

waves, electrostatic charging of the plates beneath the high-voltage cables, and the debris collected on the wires.

5.2 Power Distribution System. The power distribution system will be affected by power demand, control, and the power characteristic requirements. Cables, connectors, and grounding and bonding are areas of concern.

5.2.1 Power Cables and Connectors. Round or rectangular conductors are applicable for spacecraft systems having outputs to 50 kilowatts at 200 volts. Electrical systems with outputs of 250 kilowatts to multimewatts should consider the use of flat conductors at high currents in order to improve heat dissipation. For many Earth orbits, space charging and/or plasma shorting become serious problems at high voltage. Therefore, the distribution voltage magnitude must be limited if rectangular or sheet conductors rather than round or coaxial configuration conductors are used. Sheet conductors must be designed with heat dissipating fins or be placed on thermally cooled surfaces to radiate the i^2R losses. Conductor thickness must be considered. Very thin conductors, if subjected to high current transients or momentary shorts, would tend to sublime, permanently destroying the insulation system in the surrounding area.

Cables for large, high-power spacecraft may be required to operate at high voltage and high current, dependent upon the type and physical layout of the power source, the load power requirements, and the spacecraft orbit. The design of the power distribution system will require that the line losses be kept to an acceptable level without jeopardizing the thermal characteristics or subjecting the lines to arc-over. This assumes that high-voltage switching devices, power conversion, power transfer, transformers, and other equipment are available for protection and control of the spacecraft power generation and distribution system.

5.2.2 Grounding and Bonding. The bonding and grounding of spacecraft with power requirements of a few watts to 50 kWe will vary considerably. Smaller spacecraft, with powered loads to 10kWe will use standard, single-point grounding techniques with the solar arrays referenced to the central load module. Larger spacecraft using multiple solar array sections or other power sources, capable of being transported to space via the shuttle and assembled in space, may have a main load center and several remote load centers. Those spacecraft will require special bonding and grounding considerations.

The starting point for most spacecraft grounding designs is to utilize structure conduction only for stray voltage reference (EMI return) and for static bleed. In very few cases is the structure designed to carry power. An example of structure return is small aircraft. This avoidance of ground power currents in structure has several origins, and one is the relatively frail and lightweight spacecraft structures such that the volume current density would be too high. Also, magnesium graphite epoxies and other composites are often used, and some such as magnesium do not bond well or have low volume resistivity.

In a magnetically stabilized satellite and in any spacecraft with magnetometers on board, the structure carries no ground power since this would create an interfering magnetic field. In large satellites one finds limited use of structure for carrying electrical current. This is possible where there are relatively massive sections and largely aluminum construction. Finally, spacecraft with antennas operating in the frequency between 40 kHz and 30 MHz, must restrict the flow of power in the structure for the same reason as in magnetically stabilized satellites (noted above). For antennas operating below 50 kHz, the flow of power in structure is not acceptable because the spacecraft frame is a poor counter poise for the antenna.

Bonding to achieve a voltage reference, i.e., a "ground plane" is general practice in all metallic satellites even with a distribution system having a negative wire. This is done regardless of whether any current flow is predicted. The benefit is that there is then something to which to connect shields and bypass capacitors (filter cases). This benefit can cause problems if too much VLF to HF (audio to 30 MHz) current is injected into structure near antennas. In a satellite with a rigorous single-point-ground requirement, plus, at the same time, a rigorous bonding requirement, one has a paradox: a ground plane is created but it must not be used as a power return line. Experience shows that a good ground plane can be used for moderate level wide band returns near many antenna installations. In any case, a good ground plane will result in minimum electronics cost.

5.2.3 Circuit Interfaces. The problem that exists for large spacecraft remote centers is the voltage differential between the remote load centers and the main load center caused by line voltage drops between stations. There are three methods for connecting the communication lines between these remote control centers and the main load center; (1) hard lines with isolating transformers, (2) fiber-optics, and (3) radio frequency link.

Hard lines require that the power, communication, and control lines all be insulated from the structure surfaces and that lines other than power be sectioned and isolated with transformers. Even so, a voltage differential would exist between the ends at the line shields and structure and/or nearby power lines. Power line transients could induce large common mode voltages into the lines creating interference to the remote sensors.

Long fiber optic lines may require repeaters. Multiple connections are a problem due to the connection signal attenuation. The remote centers, however, would have their own grounds and be isolated from the main load center. Fiber optic lines are excellent choices for short lines and isolation of the sensors with respect to the load center.

Radio links require added equipment as do the fiber-optics cable system. RF interference from outside sources and transceiver reliability would be the only problem. The radio link is easiest to repair and maintain because each transceiver has an assigned location and controls a specific set of equipment on the spacecraft as shown in Figure 10.

5.2.4 Composite Structures. Graphite/epoxy and metallic/epoxy composites, when designed with correct fiber orientation, have very low or nearly zero coefficient of thermal expansion. Thus the thermal characteristics of the graphite/epoxy and metal structures can be matched to prevent cracking and reduce bonding problems. This results in spacecraft structural members which are dimensionally stable through all temperature ranges encountered on Earth and in a space environment. Figure 11 illustrates the thermal expansion characteristics of graphite/epoxy composite in the transverse direction³⁰.

Incorporation of different and necessary materials into the structure which are in direct or indirect contact with a composite can significantly affect attempts to maintain dimensional stability. Mechanically attached or bonded assemblies fabricated under or exposed to temperature changes in space are most difficult to control.³¹ Aluminum and magnesium components with relatively large coefficients of thermal expansion present the greatest problem of residual stress and structural distortion. Steel and titanium are more compatible but must be carefully designed into the structure.

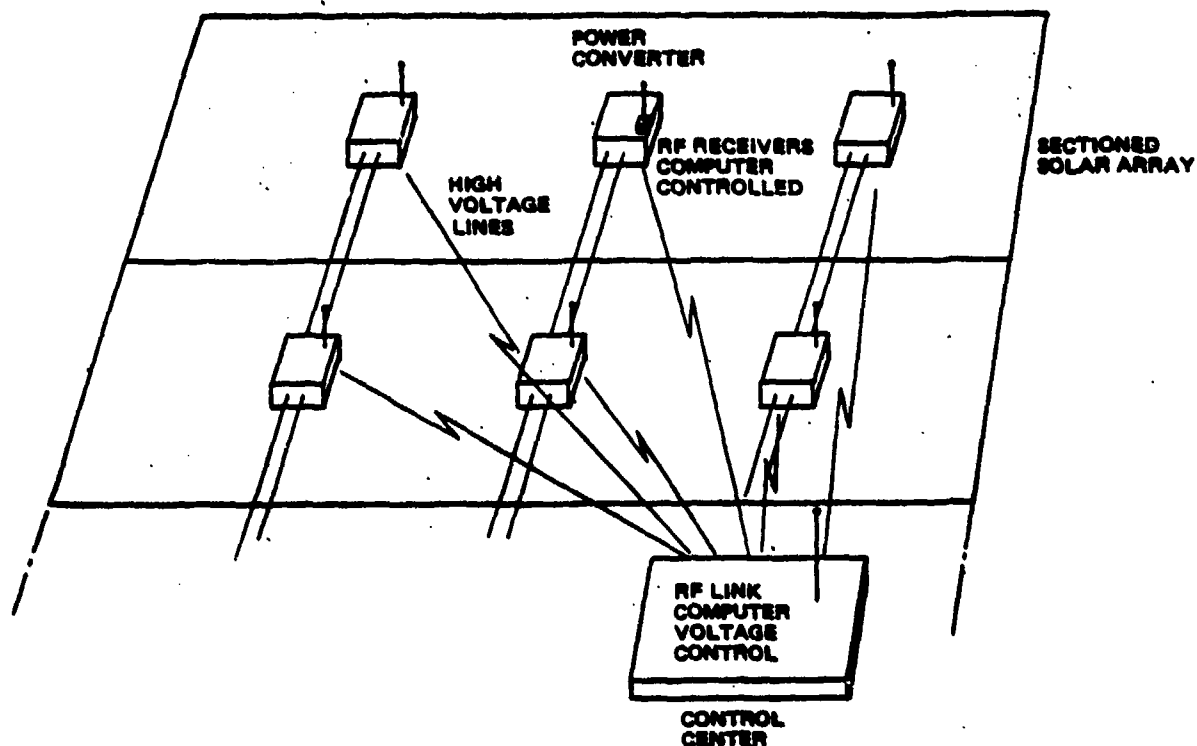


Figure 10: Spacecraft Distribution

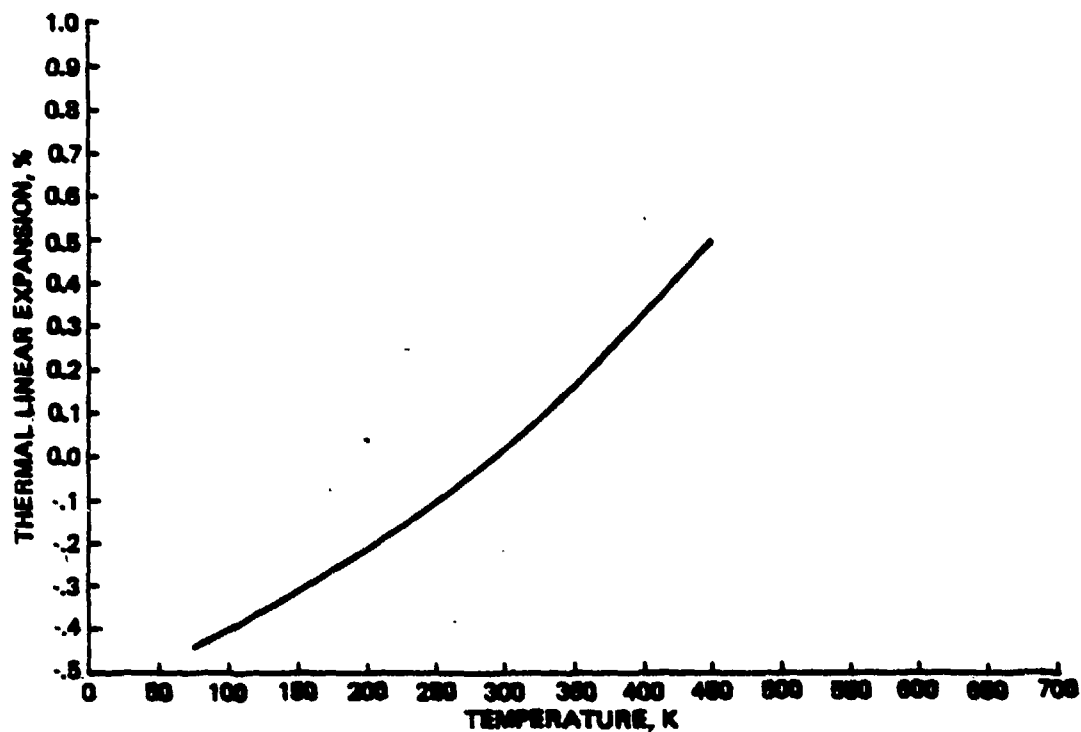


Figure 11: Transverse Thermal Linear Expansion of High Strength Graphite Fiber Epoxy Composites

Electrical conductivity of graphite/epoxy composite occurs through the graphite fibers which contact each other. Therefore, electrical contact between the composite and surrounding structure involves graphite and another (metallic) material. The electromotive potential between graphite and metal/metal alloys is sufficient to be of major concern from a corrosion standpoint. The potential difference between graphite and aluminum can theoretically reach approximately 2 volts in the presence of contaminating moisture. Structural components of dissimilar materials (prior to assembly and launch) must be maintained in a clean, sealed, moisture-free environment. Alternate protective measures involve electrical isolation of the different materials through priming and painting with conductive paints and finishes. This latter method, however, does not allow for electrical continuity.

5.3 Gases and Contaminants in the Environment. Corona and voltage breakdown of a gas are affected by temperature, charged particle radiation, ultraviolet irradiation, and particulate (space or spacecraft debris) contamination. Other contributing factors include electrode materials, electrode shape and finish, type of insulation, outgassing, electrical field stresses, electrode spacing, and applied voltage.

Contaminants can be foreign gases, dust particles, oxides, and salts. For example, helium used for leak detection, if entrapped, reduces the breakdown voltage. If mechanical or electrical stressing should cause the insulation to crack internally, the crack can fill with helium rather than nitrogen or other pressurizing gas. When the helium partial pressure is between 13.3 N/m^2 (1×10^{-1} torr) and $2.66 \times 10^3 \text{ N/m}^2$ (20 torr) it can ionize, generating partial discharges at very low voltage within the void, and possibly result in insulation failure.

Dust particles can contribute to local stress by making a plane surface have several points on the electrode. Likewise, oxides and salts, which are present in the air during the assembly, storage, transportation, and launch, can deposit on the surface of the insulation. Eventually, these deposits lower the electronic work function of the metal electrode permitting voltage breakdown at much lower voltage.

The vacuum in deep space is a good insulator because it contains few charge carriers and the mean-free path far exceeds the gap between electrodes. The volume within

most spacecraft, however, contains gas atoms and charged particles from these sources:

- a. Outgassing from nearby materials
- b. Sublimation of nearby surfaces
- c. Trapped air within the components
- d. Gas-filled voids in thermal insulation
- e. Spacecraft leakage gases

As a result, the interelectrode gap can approach the minimum pressure-spacing relationship for the initiation of partial discharges, corona, or voltage breakdown between electrodes or across a gap from the vacuum side of the Paschen Law curve. These gas sources must be minimized during design and manufacturing. However, they must be quantified so that the pressure/temperature profile in critical gaps can be calculated to permit design of a system with negligible partial discharges. Breakdown voltage for selected pure metals in uncontaminated vacuum is shown in Table 1. The breakdown voltage between contaminated electrodes may be as much as an order of magnitude lower than that between the pure metals and/or alloys of the electrodes.

Table 1: Vacuum Breakdown of Metal Electrodes

| Material (polished surfaces) | Vacuum Breakdown Voltages (kilovolts for 1 mm gap between parallel plates)) |
|---------------------------------|--|
| Steel | 122 |
| Stainless Steel | 120 |
| Nickel | 96 |
| Aluminum | 41 |
| Copper | 37 |

Unpressurized high-voltage conductors in the presence of outgassing from a spacecraft or even from high-voltage insulation are also vulnerable to corona, partial discharges, and arc-over. If there is outgassing, then the turn-on of high voltage circuits must be delayed until the pressure due to outgassing has decreased to a satisfactory low level.

5.3.1 Astronaut Suit Outgassing And Safety. When a suited astronaut performs work outside the pressurized cabin, a pressure dome around the astronaut will result. The pressure near the suit surface will rapidly decrease to 1.33×10^{-2} N/m² at one meter distance from the astronaut and to 1.33×10^{-4} N/m² at ten meters distance. This implies that all high voltage equipment, circuits, and lines must be de-energized or grounded before the astronaut enters the area. Enough time must be programmed for the system pressure to decrease to an acceptable level before the equipment is turned on, otherwise voltage breakdown will occur (figure 12). The pressure level is a function of particulate and plasma conditions at the time of turn-on. Likewise, the astronaut and his equipment must be clear of the high voltage area.

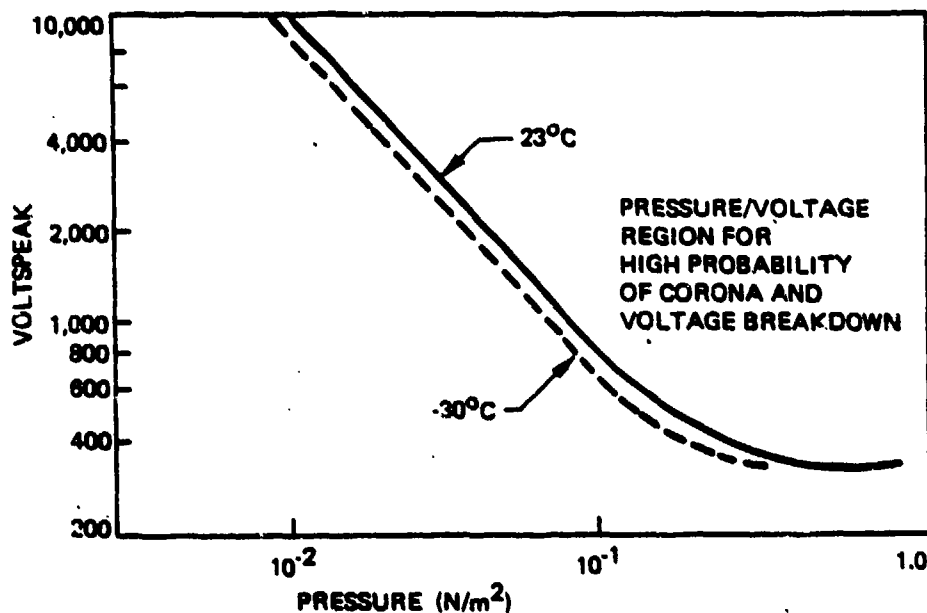


Figure 12: Paschen Curve for High Voltage Assemblies

5.3.2 Connectors And Degassing. High voltage connectors have given so much breakdown trouble that there is a tendency to not use them at all. All the junctions in a connector are soldered, with the possible exception of very high voltage points. Even then rudimentary connectors such as that shown in figure 13 are used.³² The unit requiring 15 kilovolts rests directly on the rounded central stub where the voltage is developed, thus obviating the need for leads.

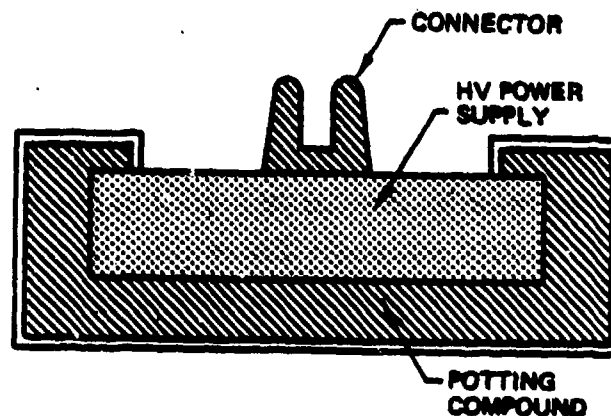


Figure 13: Illustration of Direct Contact 15 Kilovolt Assembly

Of course, some successful designers do use connectors. The key to success is adequate venting. No gas can be allowed to be trapped in the connector body in such a way that it can leak out slowly. If this happens a critical pressure environment can result for the high voltage leads in the connector long after one would have expected all gas to have diffused into space. Some connectors are vented by drilling holes through the body and insulation to permit ready egress of the gas. An adequately vented connector is the one shown in figure 14.

Figure 14 also shows a failed connector. This was a commercial connector that was simply plotted after the connection was made, with a nearly gas tight seal around the electrical leads. The gas slowly leaked out during the vacuum test. When the pressure around the leads became low enough, a breakdown occurred. The arc-over drew large currents and eventually destroyed the connector.

5.3.3 High Temperatures With High Voltage. High temperature can be very damaging to insulation. When insulation is exposed to an over-temperature (i.e., above its rated temperature) it will:

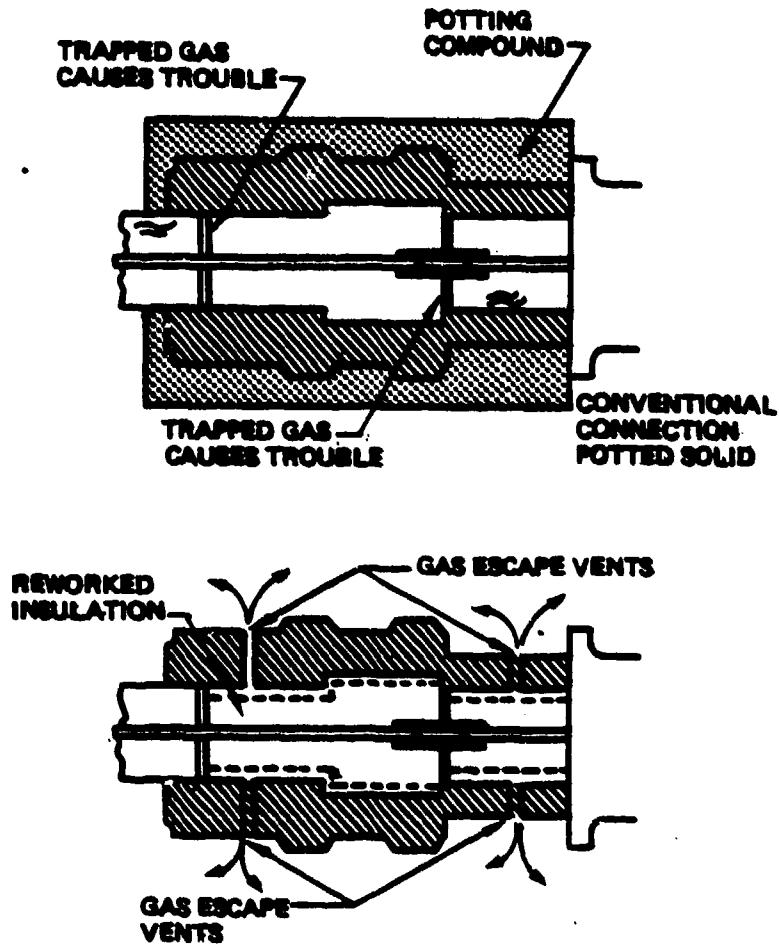


Figure 14: Vented Electrical Connectors Avoid Trapped Gas

- a. Outgas and pressurize voids and volumes.
- b. Become more conductive, which increases the probability of surface creepage and flashover.
- c. Lose part of its integrity. The insulation dielectric strength decreases as temperature increases, creating cracking and treeing and more chance of breakdown or arcing.
- d. Suffer degradation due to atomic oxygen which is more active than ozone. At temperatures above 450° F ozone dissociates into atomic oxygen.

5.3.4 Hot Spots. In the area of the highest electric field, high temperature rather than the ambient temperature, increases the quantity of partial discharges. "Hot spots" increase the outgassing rate of the insulation, thereby generating voids, minute cracks, and small pressurized enclosed volumes which enhance partial discharges and arc-over.

These "hot spots" can be the result of contaminants on the surface of the electrodes or insulation. These contaminants are things like dust particles, oxides, and salts that are present in the air during assembly, storage, transportation, and launch. The breakdown voltage between contaminated electrodes may be as much as an order of magnitude lower in voltage than that between pure metals or alloys.

These "hot spots" on bare contaminated electrodes can become hot enough (500°C) to cause thermionic emission, which would further enhance the discharge and lead to catastrophic breakdown sooner.

5.4 Materials. During the past few years designers have been deluged with new materials and processes to improve the quality and reliability of high-voltage power supplies and electronic systems. Presently, there are hundreds of stock-listed compounds and solvents to do the same job; all claiming to be the total solution, yet the failures continue to plague the systems.

Materials may be divided into two classes: repairable encapsulating materials and limited or non-repairable encapsulating or impregnating materials. Where an impregnating material is a 100 percent reactive liquid insulating material used to fill all interfaces within electrical parts and assemblies such as the spaces between very small wires on a tightly wound high-voltage transformer. Encapsulation is the embedment of small electrical assemblies within a high-voltage module or system.

5.4.1 Materials Properties. The electrical, mechanical, thermal, and chemical properties of solid insulations are discussed in Volume IV, paragraph 3.2.1 of this document. Those properties and cited references are applicable to both aircraft and spacecraft materials. Spacecraft materials have two important additional properties: outgassing and operation in near vacuum.

When operated at 150°C for 1000 hours, the thermal weight loss of a material should be less than 5%. Special materials applications which have greater weight loss or outgassing either must be enclosed or used in a module which cannot contaminate other sensitive equipment or materials within its region.

Spacecraft materials when tested per ASTM E595-77 must have less than one percent weight loss when in thermal vacuum and the condensable materials collected shall be less than 0.25 percent. Higher outgassing condensables and gases will enhance voltage breakdown within the module on nearby high-voltage equipment and the condensables will collect on critical optical and high voltage experimental plates and collectors. When testing a material in thermal vacuum, the test article must include all primers, encapsulate impregnants, and electrical and mechanical parts.

5.4.2 Materials Processing. Total impregnation or encapsulation is dependent upon the processing as well as the material properties. Some processing variables that require special attention include bare resin viscosity at the impregnating temperature, the filler material and quantity, contamination control during encapsulation curing and storage, catalyst(s) storage and compatability, working time and temperature at minimum viscosity, mold release agents, vacuum pressure cycles, cleaning materials, and residue parts materials compatability, curing, and storage (reference 33).

Proper thermal-vacuum pressure cycling within the viscosity working range of the material to eliminate voids, the proper cure and post cure cycles to reduce mechanical stresses within the encapsulating materials and across parts, and adhesion of the material to the parts are the most sought properties during processing. Voids and cracks will ultimately result in partial discharges followed by arcing and voltage breakdown.

5.4.3 Module Failures. Cracks, voids, and non-bonded surfaces account for most failures within totally encapsulated circuits. For example, applying high voltage to a capacitor will ionize the gas in the crack, heating the gas and the insulation surfaces to further increase the outgassing rate. Gas escaping from cracks alters the pressure-spacing dimension, thereby generating circuit noise. Continuing partial discharges eventually overheat the insulation, enlarge cracks, and produce ultimate voltage breakdown.

Sometimes the mechanical properties of an encapsulant are enhanced by adding fillers, usually silicon-based products. If the filled encapsulant is poured over a coil winding or over the end of a stranded wire, the filler will plug small spaces between conductors and prevent complete encapsulation of the winding. Unfortunately, the unfilled space will usually be either between windings or between the windings and metal core where partial discharges will rapidly deteriorate the remaining insulation. The outer surface of the insulation is only slightly pervious to the products of internal outgassing and will allow the internal pressure to rise toward the Paschen Law minimum for the spacing involved. At first, most voids are between 1×10^{-3} cm (0.4 mil) and 5×10^{-2} cm (20 mils). The pressure at the Paschen law minimum for these voids is between 1.0×10^4 and 2×10^3 N/m². This pressure can be sustained with continuous outgassing through infinitesimal outgassing ports.

5.5 Packaging. New packaging concepts must be analyzed and evaluated for large high-voltage spacecraft modules. Small spacecraft with loads to 5kWe have used solidly encapsulated modules, conformally coated circuit boards, gas pressurized modules, and a few liquid-filled modules. Most of the enclosed gas-filled and liquid-filled modules had volumes of less than 0.25 cubic meter. Likewise, most of the conformally coated high-voltage circuit boards operated at voltages less than 10kV. Some high-voltage circuits may require voltages of 50 to 100kV with volumes of more than one cubic meter. This will seriously change the packaging concepts and the reasons for using liquids or gases as insulating materials.

The space environment accompanies each spacecraft during its operation, but the environment is contaminated with plasma, particles, debris from space and the spacecraft, and the outgassing products from the spacecraft and resupply vehicles that will build up a sufficient pressure to cause voltage breakdown. Gas and liquid-pressurized modules have leakage and contamination problems. Liquid leaks are of the most concern because the liquid residues can collect on optical and sensitive experimental equipment surfaces. Gas leaks tend to deteriorate sealed units in time. When deteriorated, the pressurized unit is no longer useable without repressurization. This does not imply that pressure tanks cannot be built, it only means special care must be given to the seals around connectors, insulators, and covers.

5.5 Pulse Power. Pulse power conditioning is the conversion and compression of electrical energy into narrow packets of intense power released under command. Modulators or rep-rate pulsers are a class of pulse power conditioners where narrow packets of intense power are released at controlled repetitive rates (generally 1 Hz and upwards) continuously or in bursts of a finite length in time.

There is interest in increasing the peak power requirements to accommodate new high power programs like the particle accelerators, lasers, and particle beams. These new peak power requirements are orders of magnitude higher and with pulse widths an order of magnitude shorter than those used for classical radar applications; the original users of pulse power apparatus.

Multimegawatt to gigawatt pulsers are in development for terrestrial-based applications. Space systems may require multimegawatt systems in the future. Many devices developed for ground-based equipment must be modified for space. Cooling requirements, weight, and system volume constraints must be recognized. The environment and high-voltage work in opposition. Replaceable items require routine visits by spacecraft and astronauts. This will increase the gas pressure resulting from outgassing to rise sharply and, if energized, cause high-voltage units to malfunction.

5.5.1 Components. Components requiring special attention include tubes, capacitors, pulse forming networks, power supplies, and switching devices. Wire and connectors will be significant in large power systems. Transmitting tubes need better longer life cathodes and cooling techniques. Capacitors have very high electric field gradients in the dielectric next to the foil edges. The field must be reduced and controlled. Pulse forming networks must be packaged for space applications. The coils in the networks require much attention to determine spacing and fields between turns during the pulsing transients.

Power supplies can be of a high power rating and very high voltage. Tradeoffs must be considered for long life versus volume and weight constraints. Capacitors and coils with large field gradients have shorter life than low field gradient components. Switching devices and crowbar devices have been constructed for some high-voltage high-power devices. These devices must have very high reliability before they can be placed in a space system.

5.6 Multipactor. As quote from Reference 34; "Although a multipacting breakdown is a resonant phenomenon, it is not required that a unique set of operating conditions occur. There is a wide range of combinations of gap spacing, operating frequency, and voltages over which multipacting may be observed. However, there is a lower threshold voltage below which the discharge cannot occur and an upper threshold voltage above which the discharge does not occur. In addition, for a given electrode spacing, there exists a cutoff frequency below which the discharge does not occur. Multipactor discharges can occur over a wide range of frequencies and voltages above the cutoff frequency. Furthermore, in a configuration where more than one pathlength is available, less frequency selectivity would be experienced."

Multipactor breakdown may occur when secondary electrons produced at one surface are accelerated so that they reach another surface in a half-period of the applied RF field. Upon striking the electrode, the electrons create new secondary electrons which are accelerated back across the gap during the next half-period, producing more electrons when they strike. For most materials, the secondary emission yield (i.e., the ratio of emitted secondary electrons to the number of incident electrons) becomes greater than unity if the impact energy is sufficiently high. Thus, the number of electrons participating in a discharge increases when the coefficient of secondary emission is greater than 1.0 and multipacting breakdown occurs. The total number of electrons in the discharge builds up until the electrons are lost by dispersion processes as they are produced by secondary emission.

Generally speaking, the existence of a multipactor does not imply a component failure. For instance, it is possible for a multipactor to occur in a transmitting antenna in which only a small percentage of the transmitted power is dissipated in the multipactor, resulting in localized heating. The resulting decrease in radiated power and increase in voltage standing wave ratio (VSWR) may be negligible and the system may continue to operate with no other adverse effects. On the other hand, if the antenna were used in a diplexed system where transmitting and receiving occurred simultaneously on different channels, the antenna noise caused by the multipactor may blank out the receiving channel, and thus make the system partially inoperable. The total system should be considered to estimate the effects of multipacting with respect to system performance.

When designing RF equipment sufficient safety margins must be provided in the multipactor prediction to ensure that that multipactor will not occur. Frequency-temperature-pressure testing must be required to ensure product multipactor-free acceptability.

6.0 DESIGN METHODOLOGY

Each designer has techniques for developing reliable circuits, components, and equipment. The design methodology shown in this paragraph should be applied to enhance proven design techniques or aid in the development of new designs for high-voltage/high-power equipment.

6.1 Environment. All components and systems must be capable of surviving the spacecraft environment, which may suddenly change from one state to another. For example, during boost the spacecraft outside pressure changes from one Earth atmosphere at 100 kPa to less than 1 Pa in a few minutes. But the spacecraft internal environment is much different. As the spacecraft leaves the Earth's surface, the spacecraft internal and external pressures are about equal as the external pressure decreases from 100 kPa to 1000 Pa. On-board compartments containing electronic equipment outgas at a slower rate as the pressure decreases from 1000 Pa to 1 Pa. As the spacecraft enters LEO the internal pressure very slowly decreases from 1 to 0.01 Pa, a pressure suitable for some high-voltage equipment to be operating. Depending upon the location of the high-voltage modules, it may take from a few minutes for an instrument mounted on the exterior of the spacecraft to outgas to a pressure less than 0.01 Pa to over 30 days for potted, compact parts in poorly vented modules under thermal blankets as shown in Figure 2. Measured evidence of this effect is shown in Figure 1, where two component packages were placed in a vacuum chamber separated by a thermal insulation pad, the pad was held in place per spacecraft installation specification. This slow outgassing rate is of prime importance when installing unpressurized equipment in a spacecraft system.

It is important to calculate the pressure-time relationship for each high-voltage module that has exposed wiring in open construction high-voltage modules. Just as important is an estimation of the spacing so the pressure-spacing dimension can be determined. An example is the wire shown in Figures 15 and 16.

The long field lines from the furthestmost points on the wires (illustrated in Figure 15 and 16) or other conductors to a ground plane will require that a very low pressure be attained to prevent corona or voltage breakdown as shown in Figure 17.

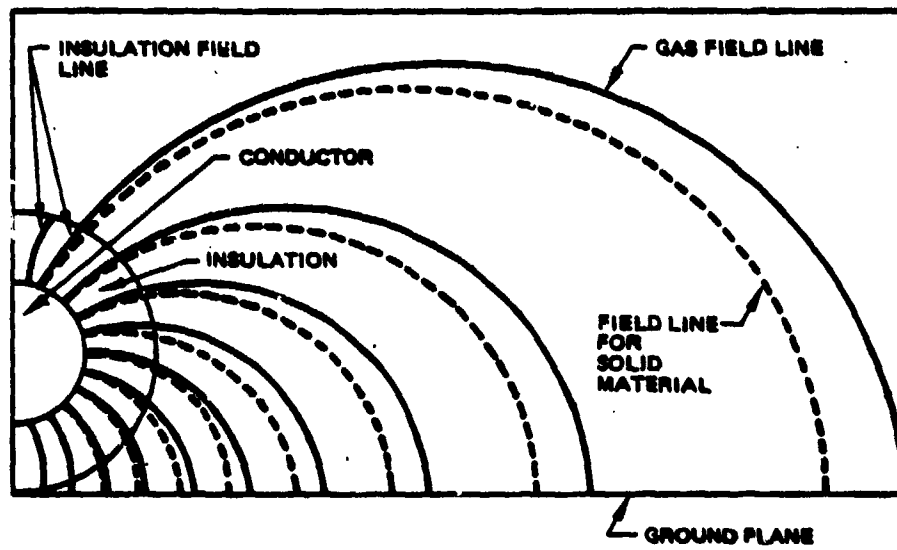


Figure 15: Field Lines Between a High Voltage Conductor and Ground

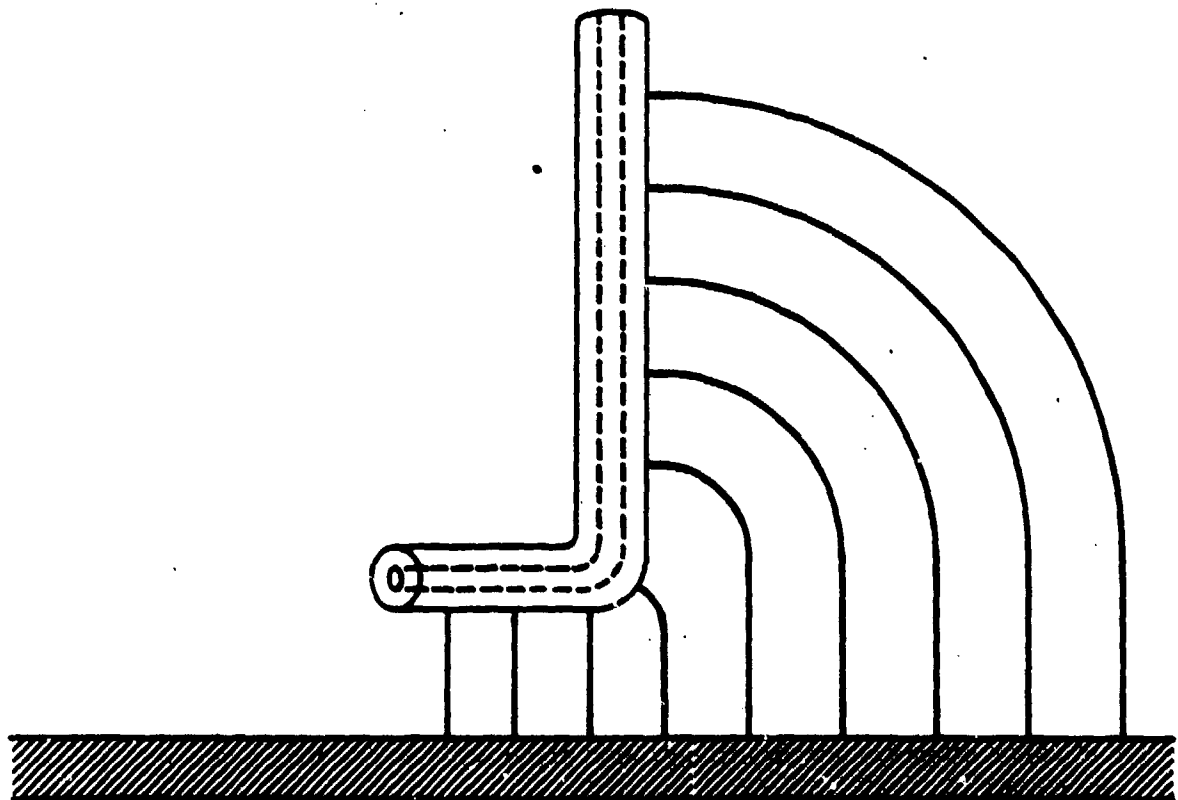


Figure 16: Electric Field Between an Insulated Conductor and Ground Plane

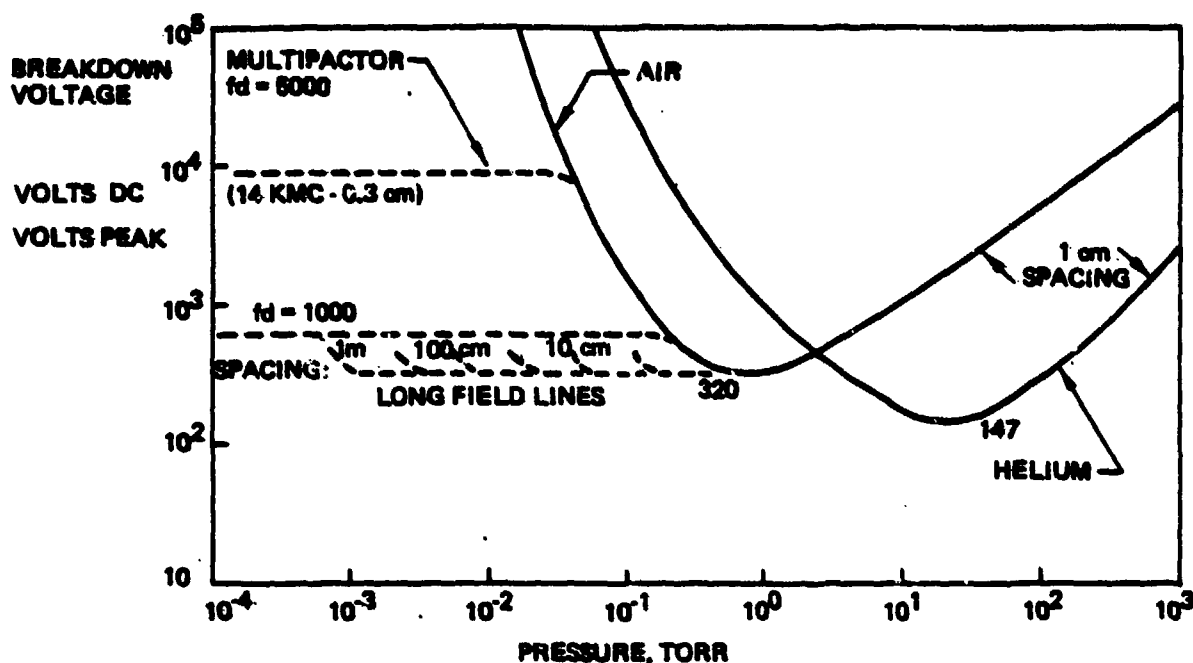


Figure 17: Theoretical Voltage Breakdown for Parallel Plate Configurations

Components within the compartments must be considered during this transition period and must be designed to withstand this pressure change. In the design the operating voltage between components and wires must include the long field paths as well as the short paths. Note the wires in Figure 15. The field between the wire and ground plane is very small or negligible, whereas the field paths from the top side around to the ground plane are long; likewise in Figure 16 where the field path from the vertical wire is long compared to the horizontal wire. This long length must be multiplied by the pressure to obtain the correct minimum pressure-spacing product to obtain the breakdown voltage using the Paschen law curve. For most designs the pressure-spacing dimension of Figure 17 can be used to obtain reasonable results. This implies that the pressure around a 10kV insulated wire or circuit board should be below 0.01 Pa before the unit is turned-on, otherwise glow discharges or gas breakdown may occur. For very long field lines the "bath tub" effect shown in Figure 18 may more closely represent the pressure-spacing-voltage requirements.

6.1.1 Exposed Equipment. Exposed equipment includes solar arrays, antennas and drive mechanisms, and modules mounted on the exterior of the spacecraft without benefit of thermal insulation. Exterior mounted equipment is influenced by plasma, and particulate.

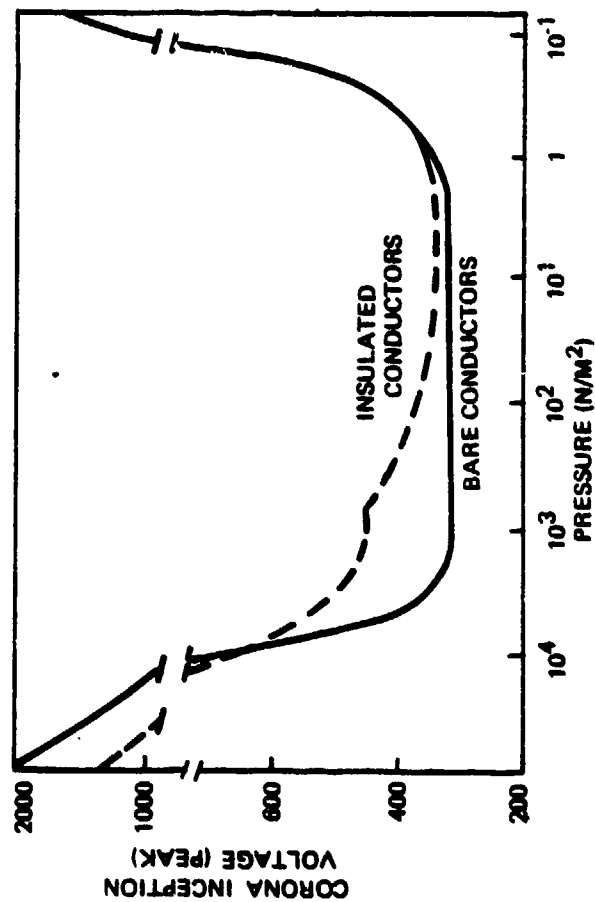


Figure 18: Critical Pressure Region for High Voltage Equipment Failure

Pressure. The gas pressure may change from that of space to 1 Pa in a very short time in the vicinity of open constructed equipment mounted on the spacecraft exterior. These pressure excursions may be caused by escape of contained gases on-board the spacecraft, spacecraft outgassing in a local region, the approach or departure of a supply spacecraft, the outgassing of an astronaut's space suit or supply lines or the exhaust from airlocks.

The voltage breakdown of air at the minimum of the Paschen law curve is 327 volts, peak. Should the gas pressure, electrode spacing product be equal to 1 Pa-cm by any combination for spacings between 0.1 and 20 cm and the voltage exceed 327 volts, peak, a glow discharge may occur. A glow discharge is not sufficient to cause breakdown but it can initiate the mechanism for a breakdown as follows:

1. First a glow discharge occurs.
2. The glow discharge will slightly heat the insulated or uninsulated surface causing further outgassing on the insulated surface.
3. Particulate on the surface will induce surface tracking and eventually breakdown.

This implies that the voltage between plasma shielded noninsulated conductors such as in the rotary joint slip rings and connections must be limited to less than 327 volts, peak. Power line transients must also be considered. These transients are at least 20%. Therefore, the line voltage plus safety margins must be less than 327 volts, peak.

or 327 Volts, peak, initiation of glow discharge
 -65 Volts, peak, transients
 -20 Volts, peak, materials heatings and electrode effects
 242 Volts, peak, maximum steady-state line voltage to assure corona-free operation in air.

Plasma. Loss of power due to plasma discharge becomes noticeable at 180 volts, peak. The higher the voltage the greater the power loss from uninsulated exposed terminals and lines. Placing a Faraday cage around the rotary joint and exposed terminations or

covering the exposed elements with conformal insulation will give acceptable operation to 240 volts, peak.

6.1.2 High-Voltage/High-Power Lines. Power supplies to 30kV have been operated in space successfully. Most of these units were low-power packages in metal modules. Large multi-kilowatt and megawatt power systems may require long power lines between the spacecraft and power source since the spacecraft will usually be large.

Cables for large high-power spacecraft may be required to operate at high voltage and high current, dependent upon the type and physical layout of the power source, the load power requirements, and the spacecraft orbit. The design of the power distribution system will require that the line losses be kept to an acceptable level without jeopardizing the thermal characteristics or subjecting the lines to arc-over. This assumes that high voltage switching devices, power conversion, power transfer, transformers, and other equipment are available for protection and control of the spacecraft power generation and distribution system.

Distribution system mass is of prime importance to the design of large high-power spacecraft. The mass of the distribution system is dependent upon 1) the lines and interconnections, and 2) the heat rejection equipment requirements (Reference 7). That weight required for distribution wiring for dc or single phase ac circuits (in grams) can be calculated by the formula:

$$W_L = 2L (A_1 D_1 + A_2 D_2)$$

Where:

D = Distribution line equivalent material density, (g/cm³)

A = distribution line Area (cross section) (Cm²)

L = distribution line length, (cm).

L = Distribution line length (cm)

A₁ = Cross sectional area phase conductor (cm²)

A₂ = Cross sectional area of the insulation (cm²)

D₁ = conductor density (g/cm³)

D₂ = Insulation density (g/cm³)

The factor 2 is included in the formula because two lines are required, a positive and a negative or return line. Since most of the larger spacecraft may be designed using

high resistivity material such as graphite epoxy, a return line is necessary (return through structure may not be feasible, nor desirable). This distribution system mass must be equal to or greater than the weight required for heat rejection.

That weight required for heat rejection for the two conductors (in grams) is given by the formula:

$$W_H = \frac{2I^2\rho L\alpha_H}{A_1}$$

Where:

I = distribution line current, (Amperes)

ρ = distribution line resistivity, (Ω -cm),

α_H - added conduction specific weight of the heat rejection system, (g/watt)

A_1 = area per unit length, (cm^2/cm).

This conductor weight is required to radiate heat so the conductor temperature can be stabilized at a given upper temperature limit.

The specific masses of the heat rejection system depend upon the shape and orientation of the conductors, the operating temperature of the conductors, and the conductor material. Cylindrical conductors present minimum surfaces for radiative cooling, whereas rectangular conductors present more surface area for cooling. For this simplified analysis it will be assumed that the conductors will be constructed of wide, thin, aluminum sheets, with the sheet longest dimension of the sheet cross-section oriented toward free space. Aluminum sheet is an excellent material because it is lightweight, has good strength and low cost, and is easy to fabricate. Other materials that may be used as conductors are shown in Table 2.

TABLE 2: Conductor Materials

| Material | Resistivity, Micro Ohm-cm (20°C) | Density g/cm ³ (20°C) |
|-------------------|--|--|
| Aluminum | 2.828 | 2.7 |
| Beryllium | 5.9 | 1.82 |
| Copper (annealed) | 1.724 | 8.89 |
| Graphite | 1589. (avg.) | 1.58 |
| Lithium | 8.55 | 0.54 |
| Magnesium | 4.6 | 1.74 |
| Silver | 1.629 | 10.5 |
| Sodium | 4.3 | 0.97 |

Several materials have better minimum weight than aluminum. Lithium and sodium have much better resistivity-density characteristics. But, these materials, along with beryllium and magnesium, are difficult to fabricate and require special storage. Copper, silver, and silver alloys are heavier than aluminum but are much easier to join and should be considered for the connector materials. A more extensive study should be made of the silver alloys and copper alloys. Some of them may have good characteristics approaching that of aluminum. Aluminum materials do tend to crack and set with time. Continual flexing (caused by electrical transients) could cause joints to weaken with time. Graphite (structure) characteristics are such that it should not be considered for the ground return; its resistivity is much higher than most metals.

Mechanical Forces. High-current transients and faults will subject the structural separators and standoffs between conductors and between conductors and ground to forces many times normal forces. These forces will tend to pull together and curl flat conductors simultaneously. In addition, large stresses will be imposed on the structural (high resistance) member. This conductor movement will flex and stretch the conductors, placing heavy tensile loads on the mated connectors and line joints.

Stacking several multiple flat conductor cables will result in a large capacitance between conductors unless all negative conductors are positioned over each other. For instance, a cable made up of several flat conductors 0.75 cm thick spaced 0.1 cm apart in a dielectric medium of $\epsilon = 3.5$ has a capacitance of approximately 35 pico-farads/meter for each pair of conductors. A 50 kWe panel could conceivably have upwards of 50 conductor pairs in parallel or a capacitance of 0.0017 microfarad/meter. This capacitance will have some storage capability to decrease the effects of transients but will add to the power source energy to do greater damage in case of a short or arc. Both conditions must be considered in the wire routing and mounting design.

Wiring in high-voltage equipment compartments should be of round conductors. For high-power and high-voltage round tubing is preferred. The tubing will hold its configuration during vibration and shock much better than insulated flexible conductors. Round tubing also has fewer problems with voltage breakdown, corona, and "clumping" or Malter effect than sharp edged square or small solid conductors.

Connector Concepts. Cables and connectors for high-voltage, high-power, high-current systems must be developed for minimum weight and efficient cooling. To

achieve these design requirements, connectors must have low contact resistance to minimize resistance and heating and have good connections with simple lightweight mechanisms. The connector design concept is impacted by the spacecraft design. For instance, much of the large spacecraft structures will be assembled in space. When the power, control, and communication wire and cables are made an integral part of the structural assembly, then all wires and cables on structural members must be mated simultaneously with the mating of structural members. Therefore, it is mandatory that the structural mating mechanism be partially engaged and the connector plate assemblies be in alignment, prior to connector engagement. Installation and connection by an astronaut is not considered a suitable design practice.

When a multiple connector plate is used, the connector plate shall mate before the individual connectors mate. Then as the spacecraft and connector plate become fully engaged, the connectors will also be fully engaged.

Connector Plate. An exploded view of the connector plate with alignment pins and the spacecraft structure docking probe-and drogue is shown in Figure 19. It is necessary that the chase spacecraft connector plate be free floating so the alignment pins can insert properly. The "free float" plate would consist of movements in all three planes. There should be some spring loading for shock absorption during final mating. This should be limited to 5 millimeters movement in the X & Y planes. The spring loading in the Z-plane (parallel to the direction of mating) will keep constant pressure on the mating connectors.

Design Concepts. The connector design concepts described below must be mechanically and electrically analyzed for applicability on a spacecraft. These concepts only partially agree with NASA and military connector specifications and standards. If they were made a part of a new, automatically assembled-in-space spacecraft design, new specifications and controls would be necessary.

Concept number one is a multi-pin, cone-shaped connector. A diagram of the connector insert and shell is shown in Figure 20. This unit is self aligning and may be connected/disconnected easily. An alignment slot on the side of the insert allows the insert to rotate up to $\pm 20^\circ$. The latching mechanism is similar to snap-lock fittings presently used and may be attached to the end of the insert end as shown, or made

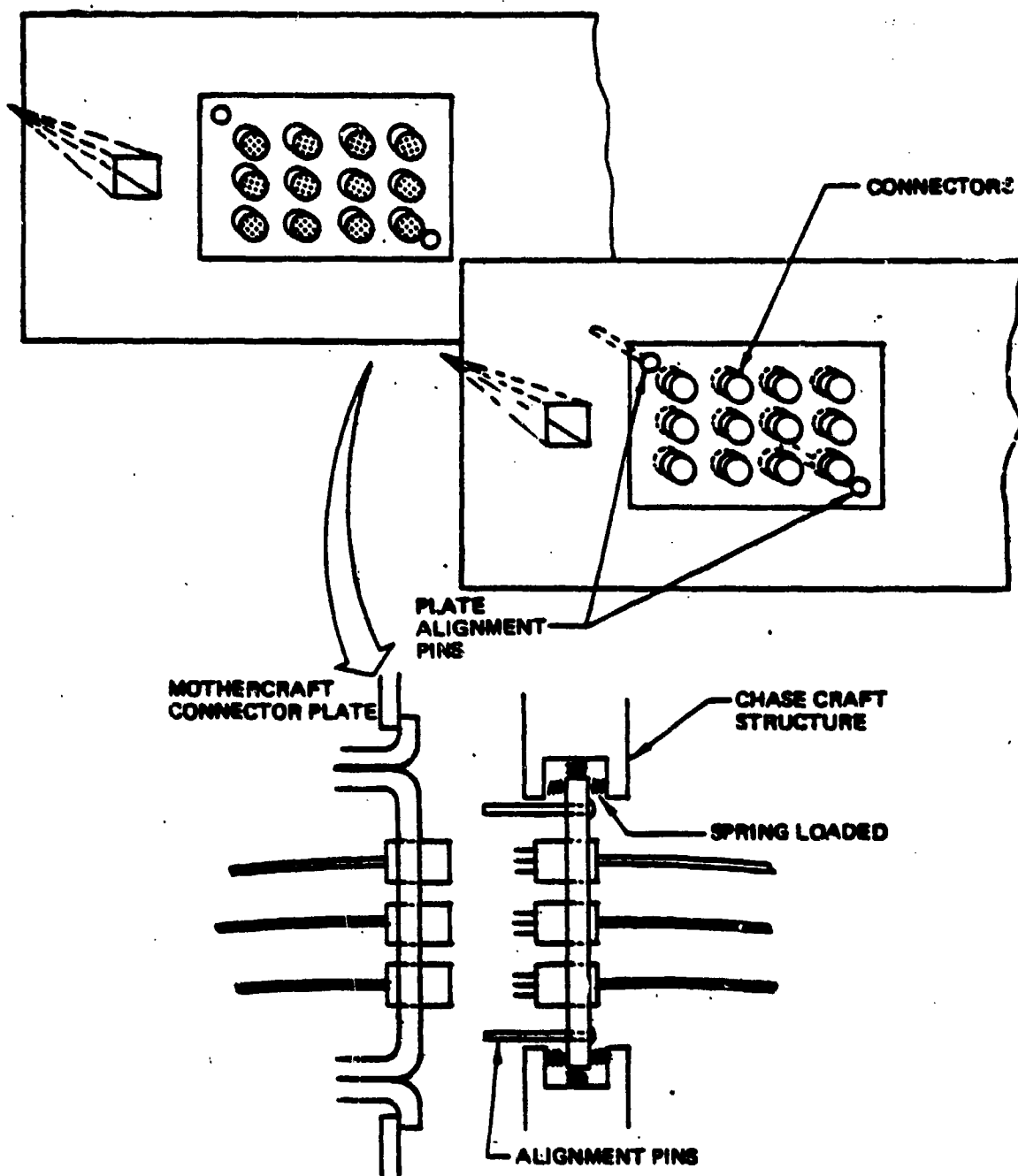


Figure 19: Connector Plate Alignment

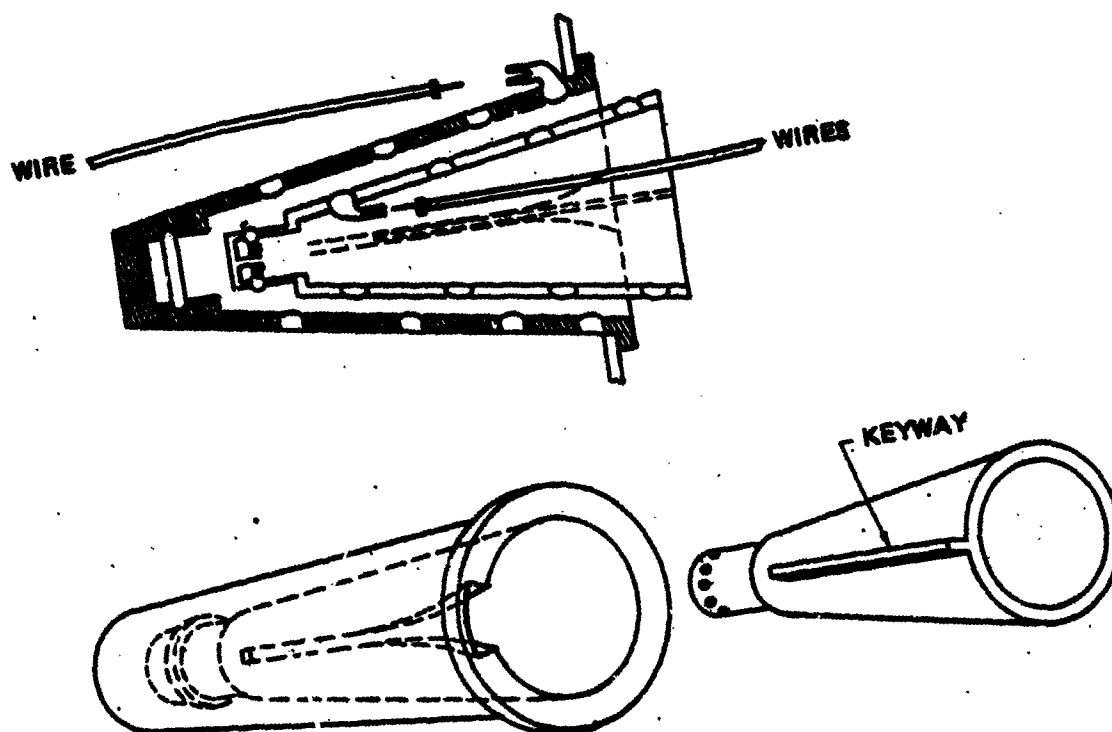


Figure 20: Multi-Pin Connector

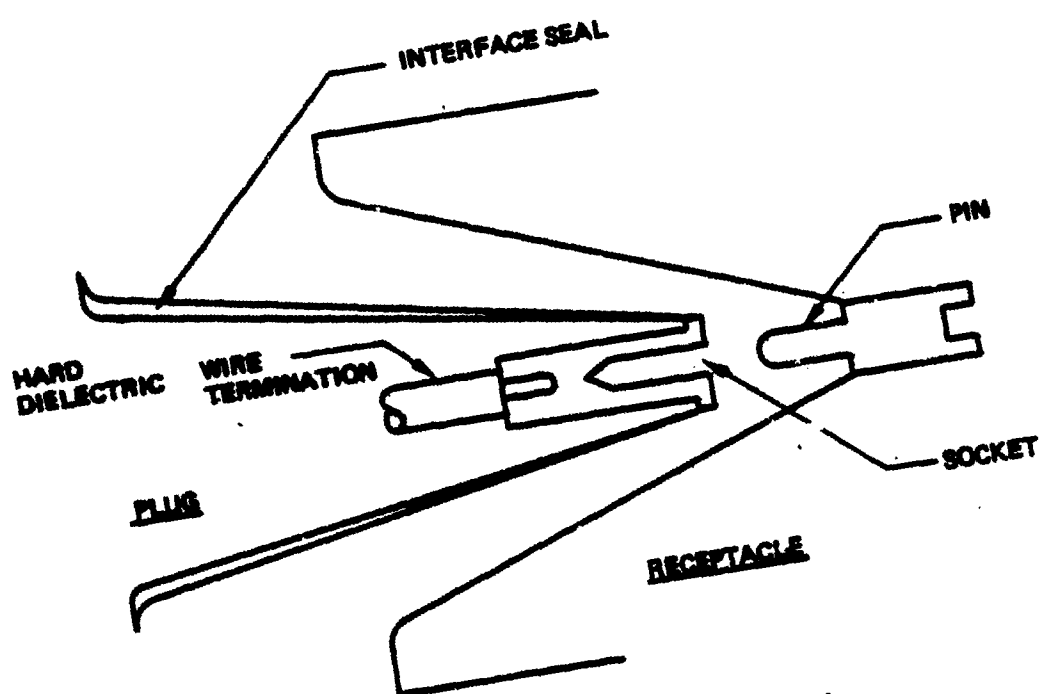


Figure 21: HV Cone Connector

part of a snap-lock ring on the large shell of the connector. All wires will be attached on the inside via insert pins to the connector contacts.

This same cone configuration could be used for single conductor coaxial configurations. In that case the coupling ring would require a spanner washer to apply positive pressure to the insert cone throughout the life of the connector. A special tool would be required to fully mate the high voltage coaxial cone type connectors. Two American manufacturers make connectors of the design shown in Figure 21.

Concept Evaluation. The connector design concepts are evaluated in Table 3 with respect to impact on spacecraft design and EVA by astronauts or mechanisms.

Environmental Effects on Wire Routing. Plasma and debris cause problems for high voltage lines, that is, open lines with over 200 volts between conductors. Plasma, being a semi-conducting medium, causes a minute current flow between the conductors, resulting in a power loss which increases with increased voltage and increased flux.

Debris, whether from space or from the spacecraft, will collect on high voltage conductors as shown in Figure 22. Debris can be collected on the high voltage conductors by: 1) direct collision of the space debris with the conductor, or 2) the electromagnetic field forcing the debris on the spacecraft surface to the conductor as shown in Figure 23. Once the debris particles have entered the field of the conductor, they will become charged and polarized to form a seta (hairlike) growth on the conductors. Some of these charged particles will form bridges or "strings-of-pearls" between high-voltage and low voltage or ground planes as shown in Figure 4. The field stress at the ends of the seta is very high, so any plasma in that region can and will initiate an arc between the conductor and ground plane on another conductor, by initiating an arc on the end of the highly stressed seta. It is important to shield the conductors and high voltage lines from debris whenever possible. The question always arises about the benefit of insulated conductors. The debris will attach to the insulated surface. For thinly insulated surfaces, less than 0.5 mm thick, the field will be greater than the dielectric breakdown strength of the insulation. Therefore, an insulation breakdown would be inevitable.

TABLE 3: CONNECTOR CONCEPT EVALUATION

| Concept Figure | Voltage, Volts | Connector Current, Amperes | Concept Evaluation | |
|-------------------|-------------------|----------------------------------|--|---|
| | | | <u>Advantages</u> | <u>Disadvantages</u> |
| 20 | 0-300V | 0-150A | <ul style="list-style-type: none"> • Lowest Cost • Lightest weight • Many standard MIL-C-38999 connectors per plate • All connectors aligned and connected simultaneously • Connectors on one plate | <ul style="list-style-type: none"> • Difficult to replace single connector • Single pin fault requires replacement of wire • Possible grounding and/or shell connection problems |
| 21 | 0-300V | 0-25V | <ul style="list-style-type: none"> • Excellent alignment • Easily manufactured | <ul style="list-style-type: none"> • Wire integral with the connector • Heavier than MIL-C-38999 • Redundant wiring required |
| 22 | 0-150kV | 0-500A | <ul style="list-style-type: none"> • Excellent alignment • Excellent shielding • Reliable - long life over 100,000 hours • Recommended for high voltage circuits where shielding is mandatory | <ul style="list-style-type: none"> • High cost • Single conductor and shield • Suitable only for high voltage |

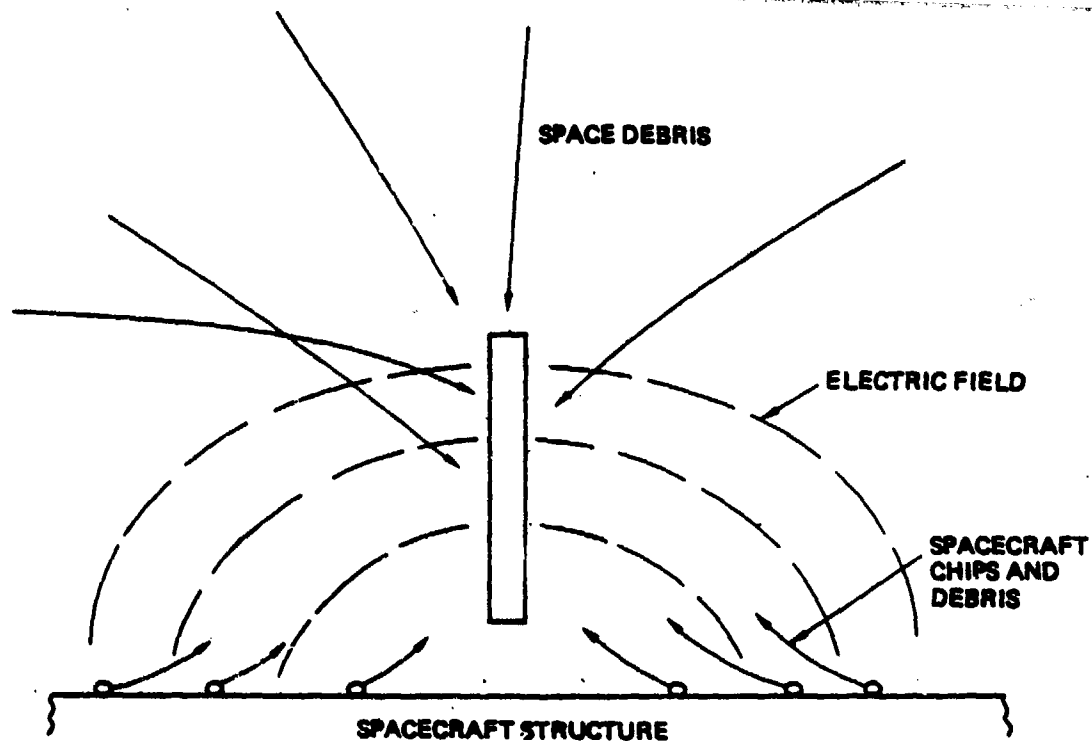


Figure 22: Particulate Collection to a High Voltage Conductor

Shielding can best be done with:

- 1) Coaxial conductors, with the high-voltage conductor inside,
- 2) Side-by-side or layered conductors to produce zero field effects,
- 3) Heavily insulated conductors.

The layered conductors, have the overall advantage since the external magnetic field is neutralized, they are inexpensive, and they are easily placed on the spacecraft. Insulated conductors will have many pin holes in the insulation which will lead to field stress concentrations at the holes, and the insulation will be subject to arcs since it will be exposed to the environment. Coaxial configurations are expensive and much more difficult to join in space.

6.2 Connectors and Connections. Electrical parts, connectors, and small components must be rigorously screened prior to assembly to assure that a circuit built of these parts is flightworthy. However, a high-voltage circuit built of screened parts (assembled and interconnected) may fail due to overstressing of the electrical insulation between the parts. In addition, electrical fields and distributed voltages surrounding the part may be much different from those test parts encountered during item screening and testing. This implies that the constructed circuit must be tested with

40 x 280 FT. PANEL
 100,000 WATT
 280 V & 400 AMP
 B ALONG THE X AXIS IS:
 B = 0.745 GAUSS
 (FOR 14 FT. APART)
 B = 0.262 GAUSS
 (FOR 40 FT. APART)

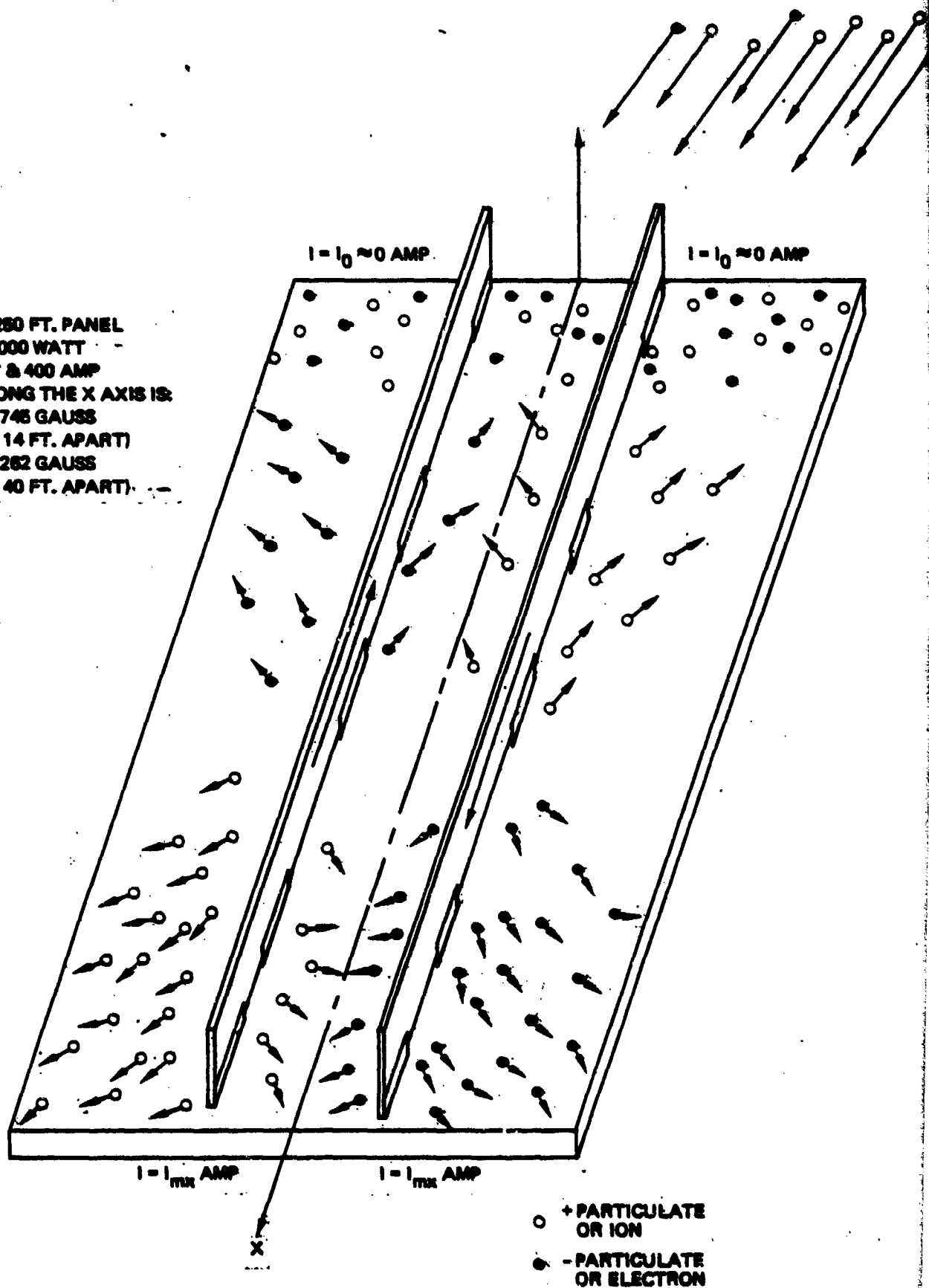


Figure 23: Schematic Presentation of Particulates - Solar Panel Interaction Under the Influence of the Magnetic Field Generated by the Flat Bus Lines

the appropriate environment to assure that the partial discharges are minimal or non-existent.

6.2.1 Wire Terminations. High-voltage interconnections between power supplies and electronic circuits are difficult to design, assemble, and assess, especially if the interconnecting shielded wire is flexed or strained after it is attached to one or both terminals. Linear stressing can break one or all of the bonded joints between the shield, conductor, or wire insulation and the terminal and encapsulating material. Frequently, teflon-insulated high-voltage wire is used. The bond between the encapsulant and teflon is weaker than that between the metal surfaces. Any delamination at the insulation bond will form a void (air gap) and result in corona and eventual voltage breakdown. When the wire is bent at the terminal (Figure 24), and the high-voltage conductor is flexed or stressed, there is greater probability of delamination at the wire-encapsulant surface.

The wire shield and insulation preparation are as important as the connection. One method of preparing the wire termination is to cut the wire and shield to length and then place three or more turns of a small conductor or formed metal ring over the end of the wire shield and insulator as shown in Figure 25. This method holds the shield end strands in place and forms a round conformal metal surface at the end of the shield. When three or more turns of wire are used, as for breadboard demonstration, the wire should be soldered in place with a low-heat iron so that the insulation will not be melted or deformed.

If teflon insulation is used, it is necessary to etch the teflon. This etching should be done within a few hours before encapsulation. If the etching material is placed on the teflon more than 24 hours at room ambient temperature before encapsulation or if the teflon is heated due to soldering, the teflon will cold-flow and ruin the usefulness of the etching. If the etching must be applied more than 24 hours before encapsulation, it is recommended that the etched surfaces be kept cool to prevent cold-flow.

Solder joints on circuit boards, especially printed circuit boards, are designed to have minimum solder. This leaves the electrical post termination protruding through the solder on the circuit interconnection. Often the post terminal is clipped after soldering, leaving sharp edges protruding above the circuit metal surface. This is unacceptable for circuits with voltage exceeding 250 volts peak. Sketches of acceptable and

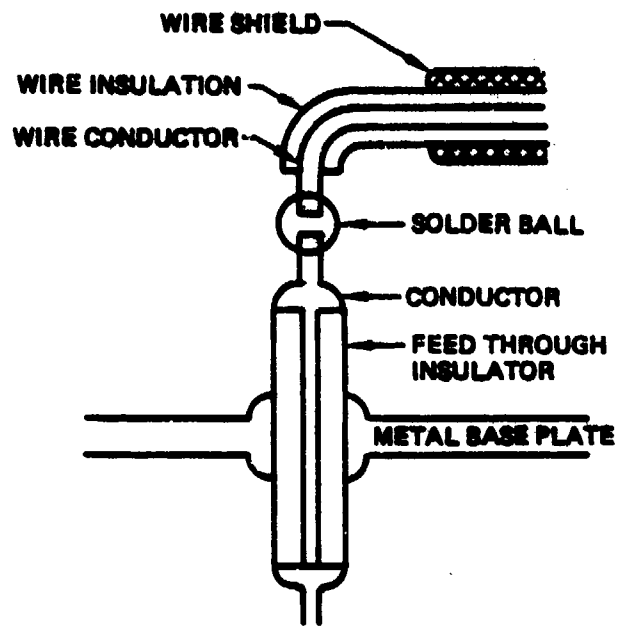


Figure 24: High Voltage Termination

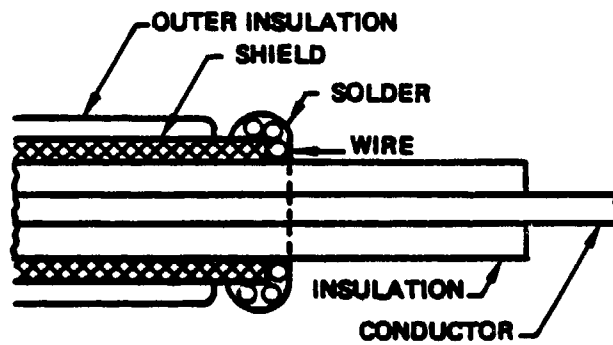


Figure 25: Wire Termination

unacceptable solder joints for high-voltage circuits are shown in Figure 26 and 27. Whenever there is a solder-draw or sharp edge protruding from a circuit, the probability of corona and voltage breakdown is enhanced. First, the conformal coating applied to the circuit board and circuits may not cover the sharp point or edges. Second, solder has more free electrons than a conformal coating. It can be assumed that a solder electrode has corona onset and breakdown of 35 percent of that of steel (this data is taken from test). Third, any ionization from a solder joint will result in surface heating of the solder, and the solder will tend to evaporate a thin layer across the insulation surface between conductors, resulting in tracking and arc over. This phenomenon will take only a few minutes between high-voltage conductors.

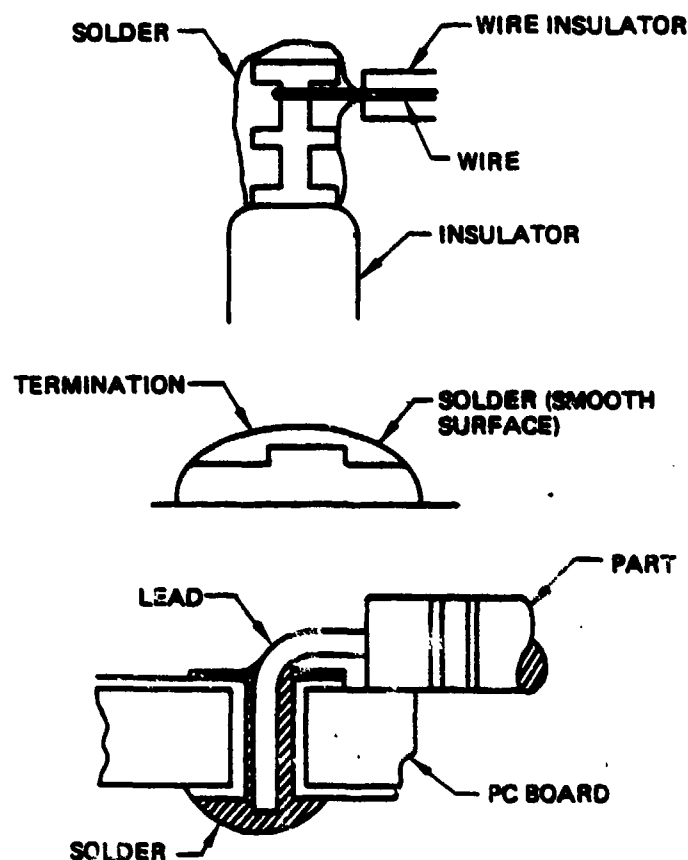


Figure 26: Acceptable Terminations

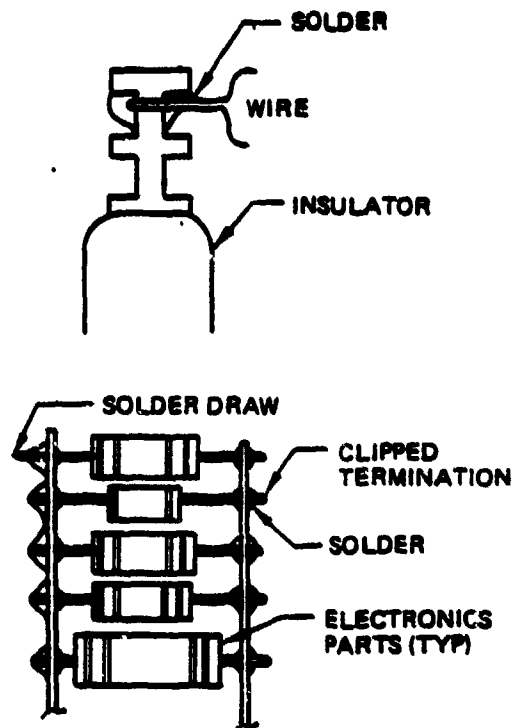


Figure 27: Unacceptable Soldered Terminations

6.2.2 Connectors. There are two basic types of high-voltage connector designs: multi-pin connectors for power and electronic equipment, and single-pin connectors between electronic packages and high-voltage power supplies.

Closely spaced, small pin, multi-pin connectors are not recommended for high-voltage unpressurized equipment, especially if there is a probability that the voltage between pins will exceed 450 volts peak in the pressure region between 50 and 0.1 Pa. This problem was demonstrated by a three-part evaluation test of a 55-pin connector. Figure 28 illustrates the pin construction and indicates the air gaps of the test article. The corona onset voltage was measured between pins of a mated 55-pin connector in a simulated high altitude chamber. The tests were performed with the connector shell electrically grounded and the connecting wires and unwired pins encapsulated with 1.25-millimeter silicone rubber on each end of the connector. In the first test, a pair of 22-gage, twisted, teflon-insulated wires (0.025mm thick) were connected to adjacent connector pins and then fed through the vacuum chamber wall to the corona test facility. The second test was like the first except that the wires were connected to nonadjacent pins. In the third test, one lead was disconnected from the assigned pin and then reconnected to an adjacent pin on the other mating half of the connector to eliminate the possibility of partial discharges between the parallel conductors.

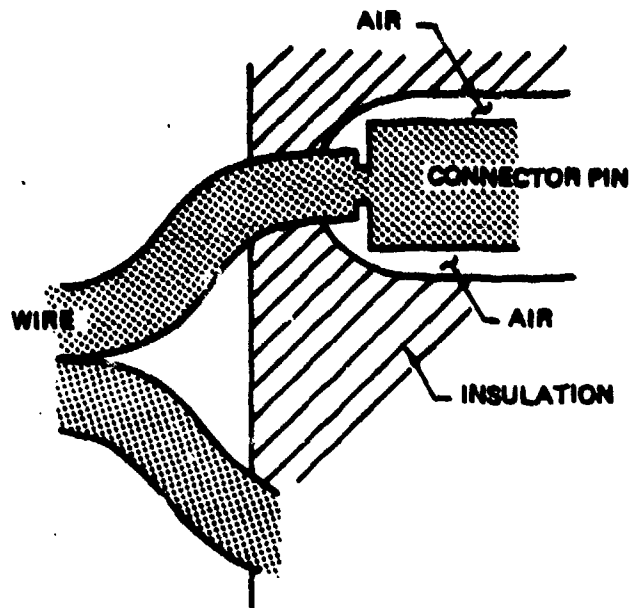
The curves drawn from the test data are shown in Figure 29. The data show that the highest corona onset voltage was derived from the test in which the pins were energized from opposite connector halves. In this test, the onset voltage for the connector was less than that of wires spaced 5 centimeters apart (the length of the connector) and greater than that of twisted closely spaced conductors. This low corona onset voltage exhibited by the connector is due to air passages within the connector construction as shown in Figure 28. This connector configuration has been designed for too low a voltage between pins to provide adequate high-voltage design practice. The curves for the other two tests have corona onset voltages equal to those of twisted insulated conductors (Reference 35).

Commercially designed, high-voltage, single-pin connectors are available for unpressurized systems. An excellent insulation design is shown in Figure 21. This design has a soft pliable insulation over the socket. The open metal surface mates with the socket metal surface, then the solid insulation bushing is forced into the pliable insulation, forcing the air from the insulation surfaces. As a desirable design feature the pin-to-shell field stress should be limited to less than 6000 volts/mm (150 volts/mil) across the insulation and the creepage path from the pin to the shell along the bushing should be limited to 3000 volts/mm. This reduces the probability of breakdown through the pliable insulation and along the insulation interface should a loose joint or small void exist due to handling and coupling.

6.3 Pulse Forming Network Components. A simple electrical pulser consists of a high-voltage dc power supply, a charging network, energy storage, and a switching device. High-power pulser may have outputs from 10 kilowatts to multi-megawatts. The higher the output power, the higher the pulse power, starting with over 10 kV to over 250 kV for larger units.

Terrestrial pulsers are designed for long life disregarding weight and volume and can take advantage of forced air or liquid cooling. Space application of pulsers must meet constrained volume, weight, and cooling limits. The requirement for special long life, high-reliability component and networks is mandatory to meet the long-life requirement.

This treatise will not include pulser designs. It will be restricted to the critical component design requirements associated with high-voltage technology.



ROUND WIRE CONNECTOR

Figure 28: Round Wire Connector

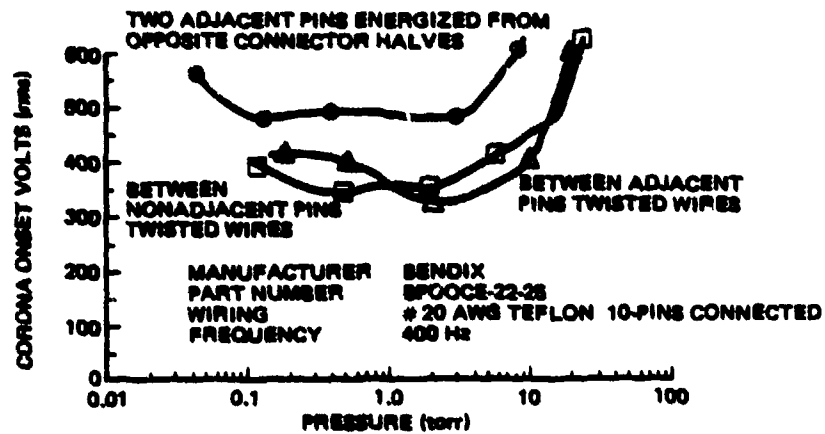


Figure 29: Connector - 55-Pins in Air at 24 °C. Terminals Not Potted

6.3.1 Capacitors. Capacitors are among the critical components in a pulse forming network. They must have low inductance, have a very rapid charge and discharge characteristics, long life of at least 10^{12} pulses with an inverse voltage rating of 20 percent. To obtain these goals, design and materials engineers are looking for better packaging, and improved field enhancement at the foil edges, as well as higher dielectric strength materials. One pulse system application has the capacitor case biased to 25 percent working voltage to relieve the stress on the outer foil and negative lead. To decrease capacitor weight some designers are working on plastic cases. Important features in capacitor design and fabrication are:

1. Obtain a void-free homogeneous film. Many thin films have upwards of 10% variation in thickness and one or more pin holes per square meter.
2. Control the foil edges. A sheared edge or square edge will have very high field stress at the corner. Depending upon the film thickness, the field at the edges of a sheared edge may be 4 to 5 times that across the body of the foil, Figure 30.
3. Eliminate dirt and debris within the capacitors during construction (rolling of the foil).

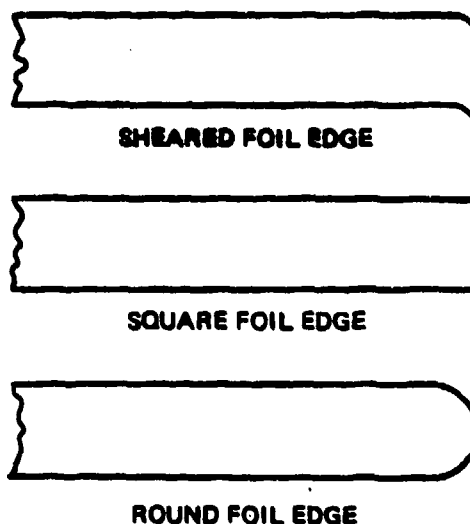


Figure 30: Capacitor Foil Edges

4. Eliminate voids between the capacitor plates or wrinkled film or foil which will enhance breakdown.
5. Use better high dielectric strength films.
6. Determine the frequency spectrum of each film material as a function of temperature. A polarization at a critical pulse frequency will cause heating and loss of the dielectric. The frequency spectrum should be measured through 100 MHz.
7. Design for low inductance. This will result from short leads.
8. Specify processes which result in an excellent bond between the foil and lead caps. A high resistance joint will result in overheating and capacitor failure.
9. Conduct trade offs between large, reliable capacitors and lightweight new designs. Doubling the number of lightweight capacitors to meet reliability values is a poor design practice when 20% more weight and volume for a single unit can achieve the same goal.

In an attempt to enhance the field at the foil edges T. E. Springer et.al.³⁶ suggests the addition of a small round wire of the same diameter as the thickness of a square ended foil. At first this appears as a logical solution to the field problem. It has the drawback of air (gas) voids existing between the foil and wire. The field at the corner would still be great enough to generate partial discharges and do damage to the insulation system, Figure 31.

High energy density capacitors have been developed and evaluated by American Manufacturers.³⁷ Films of mylar and polyvinylidene Fluoride (k-F-polymer) were used in the construction with impregnants of silicone oil, castor oil, and monoisopropyl biphenyl (MIPB); other impregnants are shown in Table 4 with the construction and design parameters shown in Table 5. Test results indicated that the k-F-polymer with silicone oil impregnant provided the best results. Edge effects were rare when the winding tension was moderate. MIPB and k-F-polymers was less attractive but have excellent radiation resistance, a feature that must not be overlooked in spacecraft applications.

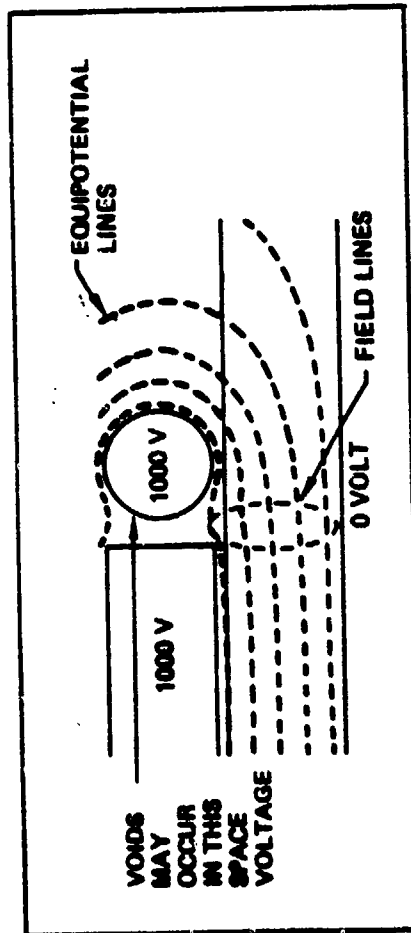


Figure 31: Equipotentials and Maximum Field for Foil With Guard Wire

TABLE 4: IMPREGNANTS SELECTED FOR SCALED CAPACITOR TESTS

| | Impregnant | k |
|----|-----------------------------------|----------|
| 1. | Tricresyl Phosphate (TCP) | 6.9 |
| 2. | Monoisopropyl Biphenyl (MIPB) | 2.5 |
| 3. | Silicone Oil (DC-200) | 3.6 |
| 4. | Diallyl Phthalate - monomer (DAP) | 10.0 |

TABLE 5: PREVIOUS DEVELOPMENT PROGRAMS ON HIGH DENSITY CAPACITORS

| <u>Capacitor</u> | <u>Construction</u> | <u>Major Design Parameters</u> |
|-------------------------|------------------------------|---|
| A | Mylar/MIPB | All film vacuum environment large temperature range |
| B | K-F/Paper/ Castor Oil | Vacuum, large temperature range |
| C | Mylar/Silicone Oil | All film |
| D | Mylar/MIPB | All film, vacuum, large temperature range |
| E | K-F polymer/ Silicone Oil | Vacuum, large temperature range |
| | K-F polymer/ MIPB | All film |

Other materials tested for high energy density capacitor use are polysulfane, polypropylene, methylene chloride, and polyethrime resin. It was found by Bullwinkel³⁸ that debris was a factor in capacitor breakdown, but the ionic impurities also play a significant role. More work must be continued in the breakdown, failure analysis, and design of materials and field enhancement to achieve high-energy density capacitors for space applications.

6.3.2 Transformers. High-voltage pulse transformers and inductors that must withstand continuous pulses have special requirements. Cast solids, liquid and films, and gas and films are used in the insulation system between turns, between windings, and between turns/windings and the core or case. The breakdown of one insulation usually results in an instantaneous breakdown of a multi-dielectric system, where breakdown results from over stressing the materials, or partial discharges. Experiments by Kao and McMath³⁹ indicate that a liquid-solid (film) dielectric system used in a pulsed voltage application has a time dependent breakdown strength in the range of 10^{-6} to 10^{-9} s. They also showed that the dielectric strength of transformer oils and n-hexane are decreased by a factor of three for an increased pulse voltage rate of rise in the range of 10^2 to 10^4 kV/ μ s.

Pulse breakdown data for gases, liquids, and solids is treated in Volume 4, "High Voltage Design Guide: Aircraft". To keep within the design limits for the pulse transformer, the maximum electric field stress must be calculated and designed so as not to exceed the average dielectric stress by a factor greater than 3.

A simple spiral wound transformer⁴⁰ is shown in Figure 32 and its equivalent circuit in Figure 33. There are two things to note about this circuit:

1. The insulation capacitance and coil inductance may form a resonant circuit if the components are incorrectly designed. This will greatly increase the field stress in the insulation circuit.
2. The capacitive circuit acts as a voltage divider when the pulse transformer has an induced pulse injected. High-energy externally induced pulses may exceed the capacity limits of initial high-voltage turns of the circuit.

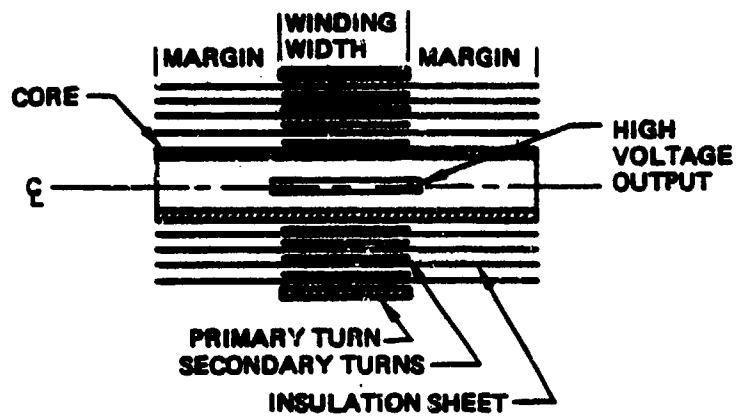


Figure 32: Simple Spiral Strip Transformer

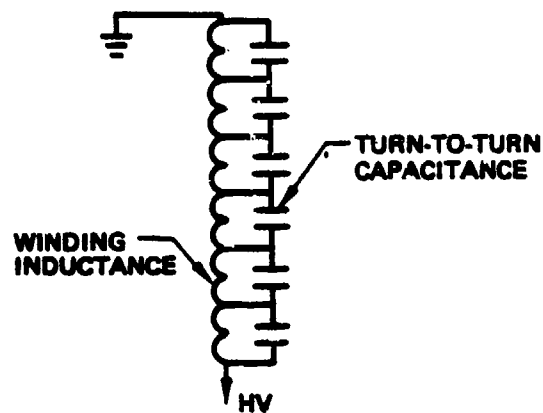


Figure 33: Equivalent Circuit Through the Thickness of a Spiral Strip Winding

Pulse transformers may be wound with either foil or wire. Square or rectangular wire and foil have small radii corners. The insulation next to these corners must be void-free and the metal free of scars, nicks, and sharp bends or wrinkles. The same precautions must be used for the pulse transformer and for capacitors. In addition, a pulse transformer may have an overshoot and/or undershoot. The insulation system must be capable of withstanding the full swing of the overshoot and undershoot.

6.3.3 Pulse Circuits. External interconnecting wires, terminals, and joints must all have corona shields. Also, the surfaces next to the high-voltage elements and parts must be free of abrasive and sharp edges on bolts, screws, and nuts.

Bushings must be designed for a maximum voltage swing. In space, solder or silver solder should not be used at the base of bushings. The solder will tend to deposit along the outside of the insulator, resulting in surface breakdown.

6.4 Packaging. Each designer has his own unique methods and techniques for compact, lightweight design. Some critical problems that the packaging engineer is responsible for include:

- Heat transfer
- Voltage stress
- Coupling between magnetic devices
- Mechanical integrity

As spacecraft power levels increase; ergo power supplies, electronic devices and packages or modules also increase in power, weight, and volume. Small, high-voltage modules to one kilowatt input may be packaged using either open construction, conformally coated parts and surfaces on the boards, or encapsulated submodules and modules. The heat can easily be directed to a cold plate and radiator. As power input to an electronic device increases, power transistors, voltage dividers, vacuum tubes and other high-power devices may have power dissipation incompatible with small cold plates installed within the module. There a fluid or gas may be better adapted for transferring the heat to a large cold plate.

In previous studies, high voltage was limited to 20kV for most space applications, with a few excursions to 30kV. Future high-power devices may involve voltages to 50kV or

more, where significant problems exist: space plasma and spacecraft pressurization to 10^{-2} Pa. System engineers must inform the packaging engineers if there is an outside chance of pressurization during the operating period, which includes the outgassing time of the high voltage equipment. If so, the high-voltage device must be totally enclosed. Likewise, high-voltage insulators are not good heat conductors and the packaging engineer must design for thermal control of the device. Finally, the mechanical integrity and spacing of parts and modules/submodules must be critically placed to withstand electrical interference and coupling and the mechanical stress of boost and orbit injection.

All large high-voltage modules rated over five kilowatts with voltages exceeding 20 kV should be analyzed for liquid or gas pressurization. To reduce hot spots and breakdown due to spacecraft pressurization, the use of gas or fluid is a good design choice.

6.4.1 Liquids. Liquids have the advantage of self circulation within the module. Also, the fluid is a natural part of magnetic insulation designs. The disadvantage is leaking and collecting on critical surfaces, thereby making other low or high-voltage equipment ineffective. Liquids must also be kept pressurized so that gas bubbles will not be formed. A gas bubble in a magnetic device will generate partial discharges and cause loss of the device. Several liquids applicable to space applications are discussed in Volume 4, "High Voltage Design Guide: Aircraft" of this report.

6.4.2 Gases. Gases must be circulated, which implies that a circulating fan must operate at all times that the system is in use. At pressures near one atmosphere, some of the better electronegative gases, SF_6 for instance, have high condensation points (-20°C) compared to the spacecraft environment. Consequently, the equipment cannot be operated until the condensed material is gasified and pressurized. Like liquids, gases also require large heavy tanks for containment. Gases, unlike liquids, will only contaminate by pressurizing the side volume of the spacecraft. Data on gases can be found in Volume 4 of this report.

6.4.3 Shielding. For very large electronic devices it will be necessary to shield the low-voltage component packages. When pulsers and other high-voltage equipment are turned "on" or "off" they produce large EMI signatures due to the fast rise/fall times of the pulse and/or partial discharges. Line filters as well as screen shields will be necessary in order to suppress the EMI.

Grounding and bonding of low-voltage and high-voltage modules and lines are important. An inadequate or high-resistance bond will subject adjacent equipment to common mode voltage excursions and EMI. Grounding and bonding is discussed in detail in Paragraph 6.7.

6.5 Insulation. There are hundreds of insulations available to select from, each claiming to solve all the insulating and encapsulating problems. Yet there are very few that can meet the rigorous requirements for space applications.

Insulation and insulation systems fall into two categories: 1) unpressurized construction and 2) pressurized construction. Unpressurized constructed modules are those that have outgassing ports which allow all parts and submodules within to eventually assume the pressure of the outside environment. The pressurized modules refers to gas-filled or liquid-filled modules. Liquid and gas-filled modules may use the insulating materials used for airplane systems provided they meet the electrical, chemical, and mechanical characteristic requirements imposed by the design.

Insulation for unpressurized systems have rigid requirements to be considered, such as:

- Outgassing
- Vacuum
- Radiation degradation
- Plasma impingement
- Ultraviolet degradation
- Temperature effects
- Shock and vibration during boost
- Long life
- High reliability

There are three general classes of encapsulating materials useable for potting high voltage electronics for spacecraft: epoxies, silicones, and polyurethanes. In addition, there are many formulations, fillers, and combinations of these basic groups used in spacecraft applications.

A survey was made of many encapsulation materials and formulations, and a great number materials were eliminated due to outgassing. NASA Johnson Space Center has issued a list of materials and their outgassing characteristics⁴¹. Before a material can

be selected, the allowable spacecraft outgassing must be known and all materials held within their respective limits. Some materials will be automatically eliminated because of outgassing.

Environment. Few organic materials can withstand the prolonged ultraviolet and/or particle radiation they would be subjected to, when mounted on the spacecraft exterior. Deterioration by UV and particle radiation would in time cause the material to crack. This would be followed by treeing and puncture by the plasma sheath.

Encapsulants within containers or shielded from the space environment will not be subjected to UV, or the space plasma. Also, particle radiation will be attenuated by the module shielding. Several high-voltage systems have successfully operated intermittently for several years in space.

6.5.1 Open Construction. Open construction refers to modules which have few or no solidly potted submodules. In this type of construction circuit boards must be conformally coated to reduce breakage due to vibration and shock, inhibit corrosion and fungus during handling and storage, and to reduce surface tracking along board surfaces between high-voltage and low-voltage parts, terminations, or connections.

Circuits selected for conformal coatings must be either limited to low voltage, installed within a module floating at high voltage, or only operate when the gas pressure surrounding the board is less than 10^{-3} Pa. Conformal coatings are put on low-voltage and high-voltage circuits in order to prevent short circuits caused by conducting debris that sometimes appears after launch. The parts are spaced so that the region outside the conformal coating is quickly evacuated when the system is placed in a vacuum chamber or as the spacecraft is boosted into orbit. Epoxy resins are the most frequently used materials for conformal coating.

Before a board is conformally coated it should be inspected. Flat surfaces mounted parallel to the board surface should be staked 0.5 to 1.0 mm above the circuit board to allow the insulating material to flow into the interspace and fill the void between the part and the board. Then the following rules should apply:

- a. All boards, conductors, wiring, and electrical components must be cleaned per the appropriate specification before the unit is conformally coated. This includes solder flux, finger prints, and particles from the work bench and dust.
- b. The circuit boards should be conformally coated with at least 3 separate layers of a low viscosity insulation. Application may be either by dipping or brushing with each layer applied at right angles to the preceding layer. Three layers are recommended to eliminate the pinholes (continuous leakage path) and uncoated areas that normally occur in single or double coating processes. The completed process should be checked by an insulation test.
- c. The final step in an electrical assembly is the joining of the printed circuit board assembly. If wired, the wire and solder joints must be cleaned and conformally coated with the same precaution as the electrical networks on the printed circuit boards after the connections are made to the other boards subassemblies within the module.

In many cases a conformal coated board is then potted to take advantage of the strong points of both methods. When so doing the board and assembly parts must be cleaned and care must be taken to insure good bonding between the two substances.

Some materials that are used for conformal coatings include:

Conathane CE 1155

Uralane 5750

Two materials that require a post-cure vacuum bake to meet the outgassing specification are:

Hysol PC 18M

Solathane 113

Parylene is an excellent coating material; its drawbacks are poor adhesion to other materials and repairability.

Printed Circuit Boards. High-voltage printed circuit boards must be designed for high-voltage. High-voltage terminals may be mounted on kel-F (a fluorocarbon material) stakes or epoxy molded standoffs to separate the voltages such as shown in Figure 34. The board must also be designed for high-voltage. The surfaces may form creepage paths as a function of time. These creepage paths can be inhibited by cutting slots in the board to break up the fields. A field plot is recommended to determine the optimum placement of the slots. Note that the slots may be curved or L-shaped to inhibit some field paths. Board selection is very important. Fiberglass resin-filled boards are used extensively. Some of the boards tend to delaminate when subjected to excessive pressure. Creepage paths will grow within the cracks and cause a board/circuit failure. Some boards have been made of velspar and other solid materials with success³. The advantage of this construction is that these materials can be machined to increase creepage paths. In addition, their surface resistivities are very high. Conformal coating materials must be evaluated for compatibility with each material used for construction and for parts coatings.

6.5.2 Encapsulation. One method of preventing a gas discharge voltage breakdown is to exclude gases from the high-voltage areas. This can be accomplished by encapsulating the high-voltage circuitry. Encapsulation provides the system with mechanical protection from external damage, gives structural support to the components against shock and vibration, and protects the high-voltage system from gas discharge damage.

The decision to encapsulate should be made during the initial design concept phase and incorporated in the subsequent hardware design. In this manner, a total system approach to the design can be taken, yielding a power supply with minimum problems that can arise from encapsulation or potting. This will permit the optimum choice of components, parts, materials, mechanical arrangements, manufacturing techniques, and the methods of function and environmental testing. This approach will reduce the failure probability. In a survey of U.S. Air Force, NASA, and industry, it was found that encapsulated electronics failures fall into four categories: high electric stress (packaging); selection of parts or improper use of parts (design); prior bonds, board delamination (material); and voids, cracks, workmanship, and handling (processes). The evaluation of these items are summarized in Table 6.4¹

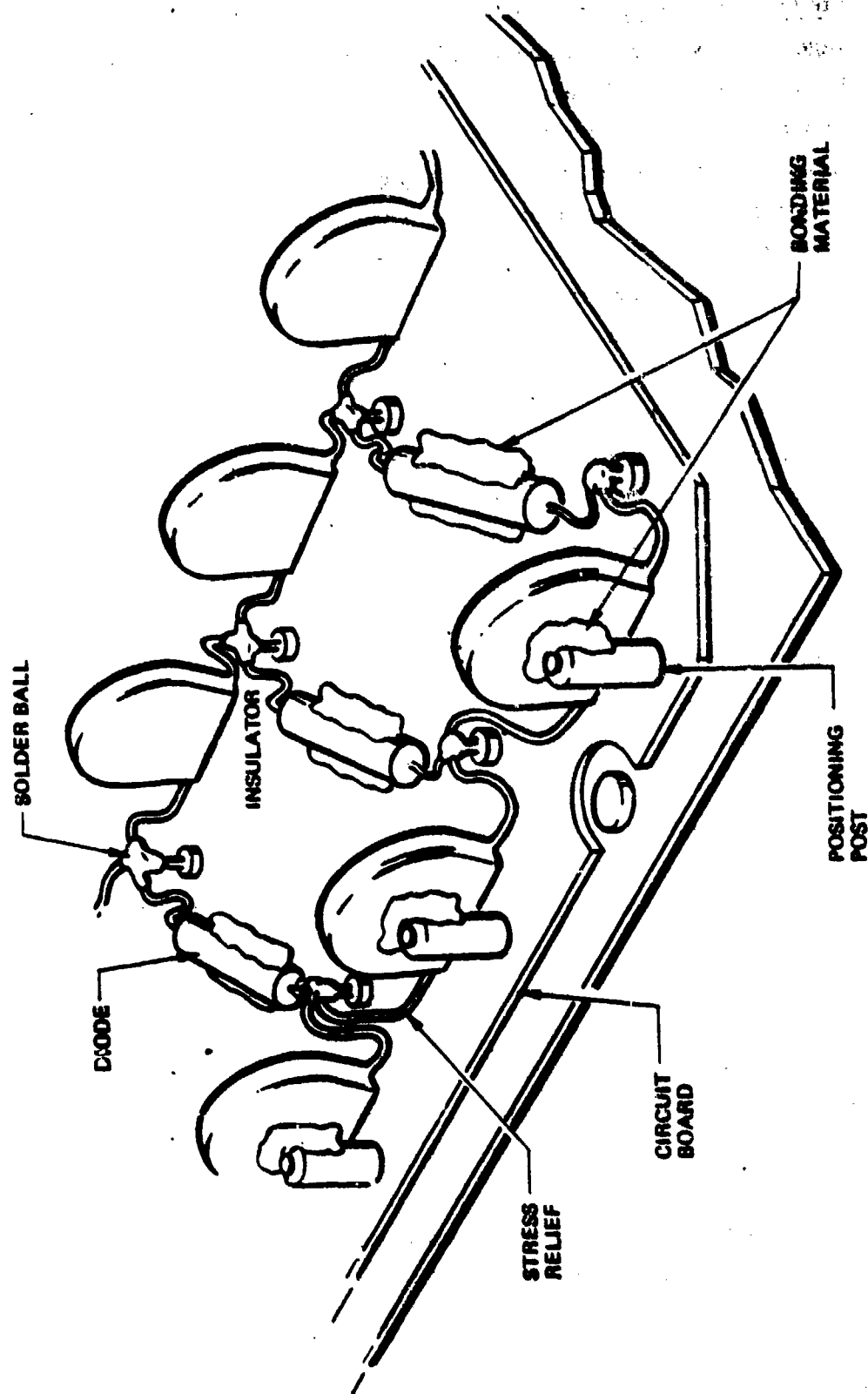


Figure 34: High Voltage Circuit Board

TABLE 6: SUMMARY OF FAILURE ANALYSES FROM SURVEY

| <u>ITEM</u> | <u>PERCENT</u> |
|-------------|----------------|
| Materials | 15 |
| Packaging | 24 |
| Design | 25 |
| Processes | 36 |

It should be noted that selecting the encapsulating material and designing the circuit and packaging can be accomplished with reasonable success. Processing the materials gives much more trouble.

6.5.3 Processing. Processing involves everything from cleaning the circuitry to be potted to mixing, pouring, curing, and post-curing the encapsulant.

Cleaning the molds and circuits to be potted is critical. Some rules to follow are given below³³:

- o Work tables to be used in this process shall be covered with a clean plastic or paper cover during assembly of the component into the mold.
- o New molds shall be solvent washed and vapor degreased in methyl chloroform to remove oils and other foreign materials. Heat the mold to $168 \pm 8^{\circ}\text{C}$ and apply a generous coating of carnauba wax. Bake the mold for two hours at $162 \pm 8^{\circ}$.
- o Spacers molded of the same material or a material compatible with that used in the encapsulated assembly will be used frequently to maintain required dimensions between the components and the mold faces. These spacers shall be cleaned with inhibited methyl chloroform.
- o All visible flashing and contamination shall be removed from recycled molds by brushing and scraping. Use only wooden, plastic, or brass scrapers. Care must be taken to avoid scratching or nicking molds during flash removal.

- o The unit to be assembled into the mold shall be handled carefully to avoid damage to the components in the unit or to lead wires, and to prevent introduction of any foreign material, mold release, or other contaminants.
- o Adhesive tape shall be applied to the mold parting lines. If tape does not provide a complete seal, a coating of silicone rubber shall be applied.
- o Teflon surfaces shall be treated with Teflon etch solution and the component assembly to be impregnated shall be vapor degreased. The mold and component assembly shall be prebaked at $135 \pm 50^\circ\text{C}$ to remove all moisture in accordance with the following schedule:

| <u>COMPONENT ASSEMBLY WEIGHT</u> | <u>BAKE TIME</u> |
|----------------------------------|--------------------|
| 0.25 pounds and less | 2 to 2 1/2 hours |
| 0.26 to 5 pounds | 4 to 4 1/2 hours |
| 5.1 to 25 pounds | 8 to 8 1/2 hours |
| Greater than 25 pounds | 12 to 12 1/2 hours |

- o The mold and assembly then shall be transferred to another oven and baked at $70 \pm 50^\circ\text{C}$ for 2 1/4 hours. Transfer the assembly immediately to the vacuum potter.
- o The assembled molds shall be evacuated to a pressure of 0.5 millimeters of mercury or less, and maintained at that pressure for a minimum of 30 minutes.

Parts and Circuits Cleaning. Scrub solder joints and other areas that are contaminated with hard-to-remove deposits of foreign matter using a stiff, short bristle brush wetted with isopropyl alcohol. Then thoroughly clean all areas to be potted with fresh solvent. Allow the solvent to completely evaporate before proceeding with the next operation.

Vacuum Impregnation. The basic and foremost objectives during impregnation of parts is to replace all cavities with the encapsulant and have a completely void-free, crack-free material. In order to achieve maximum filling, it is usually necessary to reduce the encapsulant viscosity. The application of heat reduces viscosity, but the reaction

rate is also increased, thereby shortening the potlife, in order to determine the optimum minimum viscosity, the mold and part temperature can be increased to a value determined by experiment. Heating the parts is preferred instead of heating either the as-mixed material or the pre-mixed components where the degree of cure is more difficult to control.

Potting. As soon as the encapsulant is mixed, it should be placed in a vacuum and outgassed to remove all extraneous gases. Then the material can either be poured into the mold and the mold evacuated followed by pressurization or it can be siphoned into the mold inside the vacuum chamber. The potting material is drawn by the vacuum inside the chamber by opening a clamp in the siphon that is immersed into the encapsulant. The vacuum must be maintained at all times during this process to eliminate gases.

Removal of trapped air is mandatory for producing void-free potting. Either vacuum exposure after liquid filling or liquid flow-through molding can be used. Vacuum molding is less sophisticated in that just a simple box mold can be used. Flow-through molding requires critical tooling for proper location of inlet and exit openings, proper pressure application, and usually requires some sort of automated meter mixing.

The time of vacuum exposure is critical since too short a time will leave entrapped voids and too long a time may remove catalyst and alter the properties of the cured material or might possibly prevent cure.

Over pressurization with dry nitrogen is used after evacuation to promote flow and filling of potting material into deeply buried voids. Too little pressure would be ineffective and too much pressure could force nitrogen back to cause some void areas.

There is an optimum time for the application of overpressure. Too short a time will not provide enough flow time for proper filling. Too long an overpressure time is inefficient use of time.

The above four process variables should be selected because of their importance in completely filling densely wound coils. Other important process parameters which must be held constant include:

Dip priming
Primer cure time and humidity
Resin cure time

Each material has its own unique cure time and temperature requirements. These values must be honored for best results.

After the assembly is removed from the mold and inspected for blisters and voids, it should be post-baked to relieve mechanical stresses within the encapsulant. Post-baking may require a temperature to 150°C for 96 hours to meet all outgassing requirements.

6.5.4 Encapsulant Selection. There are three general classes of encapsulants, potting materials, or conformal coating materials which are generally acceptable for spacecraft uses: (1) epoxies, (2) silicones, and (3) polyurethanes. The main characteristic of selections from these three polymer types is their low outgassing behavior, which reduces the problems of spacecraft contamination and internal spacecraft pressure conducive to electrical discharge.

Properties of interest that must be considered in the encapsulant selection are shown in Table 7. Of these properties, some are more important than others. They are: (1) sufficiently low viscosity and sufficiently long pot life to give the highest potential for high yield, void-free encapsulation, thereby best assuring freedom from corona problems, (2) good thermal stability in terms of service temperature, weight loss, service life, etc., so as to minimize polymer degradation and outgassing in system use, (3) low thermal expansion so as to minimize system performance failures due to differential thermal expansion between normally low thermal expansion component parts and normally high thermal expansion encapsulating materials, (4) high track resistance so as to minimize potential of carbon path formation in arcing conditions which usually leads to early catastrophic system failure, and (5) good overall electrical characteristics, such as high dielectric strength, high resistivity, and low dielectric constant and dissipation factor. The best combination of these desirable characteristics is likely to give highest yield, highest reliability, encapsulated high voltage electronic modules. Some material properties that can be considered are listed in Table 8.

TABLE 7

PROPERTIES OF INTEREST FOR INSULATING MATERIALS

| <u>MECHANICAL PROPERTIES</u> | <u>ELECTRICAL PROPERTIES</u> | <u>THERMAL PROPERTIES</u> | <u>CHEMICAL PROPERTIES</u> | <u>MISCELLANEOUS PROPERTIES</u> |
|---|--|--|---------------------------------------|----------------------------------|
| Tensile, compressive, shearing, and bending strengths | Electric strength | Thermal conductivity | Resistance to reagents | Specific gravity |
| Elastic moduli | Surface breakdown strength | Thermal expansion | Effect upon adjacent materials | Refractive index |
| Hardness | Liability to track | Primary creep | Electro-chemical stability | Transparency |
| Impact and tearing strengths | Volume and surface resistivities | Plastic flow | Stability against aging and oxidation | Color |
| Viscosity | Permittivity | Thermal decomposition, Spark, arc, and flame resistances | Solubility | Porosity |
| Extensibility | Loss tangent | Temperature coefficients of other properties | Solvent crazing | Permeability to gases and vapors |
| Flexibility | Insulation resistance | Melting point | | Moisture Adsorption |
| Machinability | Frequency coefficients of other properties | Four point | | Surface adsorption of water |
| Fatigue | | Vapor pressure | | Resistance to fungus |
| Resistance to abrasion | | | | Resistance to aging by light |
| Stress crazing | | | | |

TABLE 8. DESIRED MATERIAL PROPERTIES FOR SPACECRAFT APPLICATIONS

| ELECTRICAL | | MECHANICAL | |
|---------------------|--------------------------|----------------------------------|------------------------|
| PROPERTY | VALUE | PROPERTY | VALUE |
| ARC RESISTANCE | >60 Sec | SHRINKAGE | |
| DIELECTRIC CONSTANT | ≥6 | VOLUME | <3% |
| DIELECTRIC STRENGTH | >350V/MIL | LINEAR | <3% |
| | | AGE | <0.5% |
| | | SERVICE TEMP. | -55° + 105°C |
| SURFACE RESISTIVITY | ≥10 ¹² OHMS | HEAT DISTORTION (TEMPERATURE) | >100°C |
| VOLUME RESISTIVITY | ≥10 ¹² OHM-CM | | |
| MOISTURE ABSORPTION | <1% | COEF. OF THERMAL EXPANSION | <1.5 x 10 ⁴ |
| FUNGUS | NON NUTRIENT | | |

Viscosity. The importance of low viscosity for encapsulating materials cannot be over emphasized. Viscosity relates to the ability of the encapsulating material to thoroughly fill all of the interstices in high voltage assemblies, and to thoroughly impregnate critical, tightly wound coil devices such as transformers, the wells around connector pins, and the small gaps between the parts and boards on printed circuit boards. The use of low viscosity encapsulating resins simplifies vacuum-pressure processing which is one of the critical factors in achieving a high yield, high reliability encapsulated, high-voltage electronic assembly.

Pot Life. Pot life is another important encapsulating material characteristic. Pot life is the time during which the encapsulating material retains a low viscosity during the encapsulation process. Thus, the considerations with respect to internal voids apply to pot life as well as to viscosity. It is desirable to have a long pot life, that is, to retain low viscosity during the complete processing cycle. If the encapsulating material gets thick too quickly, voids are produced during the encapsulation process.

Adhesion of Encapsulating Materials to Components. This characteristic of encapsulating materials is also important for encapsulation of high-voltage electronics. Lack of adhesion leads to air voids, thereby allowing corona or arcing and tracking problems. Epoxy encapsulating materials have excellent adhesion properties. Silicone encapsulating materials do not have good adhesion alone, but can be made to exhibit good adhesion to varying degrees when suitable primers are applied to components in a controlled process.

6.5.5 Material Used in Spacecraft Systems. Encapsulating materials that have demonstrated good performance in spacecraft applications are shown in Table 9. Each of these materials has its limitations and must be evaluated for operating temperature range, thermal coefficient of expansion, and compatibility with parts within the module.

A quick literature survey will also reveal many products that appear right for the application because some reliable manufacturers have used the material. Often the user of a material will modify the formulation to suit his needs. When this is done it may change one or more of the properties of the original mixture. When a user has success with a specific formulation it may be placed on the market. Such materials are shown in Table 9.

TABLE 9

PERFORMANCE OF INSULATING MATERIALS AT LOW
TEMPERATURES AND 10^{-4} N/cm² PRESSURE IN
SPACECRAFT APPLICATIONS

| <u>Material</u> | <u>Working Voltage</u> | <u>Temperature</u> | <u>Comments</u> |
|------------------------------------|----------------------------|--------------------|--------------------------------|
| Conap 2521 | 3000 Vrms | -55 to 125°C | No damage |
| Solithane 113/300 Formula 12 | 15 KV | -40 to 85°C | Successful |
| Scotchcast 280/281 | 3000 Vrms | -40 to 125°C | Successful for Transformers |
| RTV 615 | 20 KV | -55 to 85°C | Successful |
| 6154 | 15 KV | -20 to 85°C | Successful |
| Adriprene | 5 KV | -40 to 85°C | Successful |
| PR 1538 | 5 KV | -40 to 85°C | Successful |
| DC 93-500 | 15 KV | -40 to 85°C | Successful |
| RTV 3140 | 1.5 KV | -40 to 85°C | No Damage |
| RTV 11 | 4.5 KV | -40 to 85°C | Successful |
| Formulated Epoxy | 10 KV | -40 to 85°C | Successful |

6.5.6 Materials Evaluation. Data sheets showing the electrical, chemical, and mechanical properties for each material are usually limited to ASTM requirements. They do not include variance in mixing, shelf life (storage), and processing. For example, addition of filler varies some of the properties considerably.

Concentric-Cylinder Test. The concentric-cylinder test is a nonstandard ASTM test and may be inappropriate for many potting applications. The test was designed to evaluate and simulate thermal-mechanical stresses within high-voltage potting compounds confined between rigid surfaces, such as traveling wave tubes, transformer windings, cores and bobbins, and parallel metallic or ceramic surfaces⁴². When a material fails to pass this rigorous test, it does not imply that the material is unacceptable as a high-voltage potting material. It only suggests that a few material applications are too restrictive for all materials.

The test cell is constructed from two concentric aluminum tubes. The outer tube has a 1-3/4-in. inside diameter and the centered inner tube has a 1-inch outside diameter. These tubes are set up vertically on a flat surface and filled with potting compound. A photograph of the test cell parts and the assembled test cell is shown in Figure 35. The test cell determines the adhesion properties with concave and convex surfaces and whether the shrinkage of the material on curing and testing will break the adhesion. A potting material that wets well, a desirable characteristic, will have a concave meniscus at the top surface. It was found necessary to provide a similar configuration at the bottom surface, as shown in the figure, to avoid a stress concentration at this point during thermal shock testing. The necessity for stress relief by avoiding sharp inside corners should be applied to the design of the power supply enclosure.

Thermal Shock Adhesion Tests. Testing is done by placing the cured test material and test cell alternately in an oven and a cold box. A test condition of 2 hours at high temperature followed by 2 hours at low temperature, with less than 1 minute transfer time between temperature extremes is good for thermal shock evaluation. The material fails if it cracks or unbonds from the aluminum tubing at any location. Evaluation for cracks and voids is made by testing each test sample before and after each thermal-shock test. The cylinder and material must be cleaned and dried before partial discharge testing to remove residues accumulated by handling and humidity.

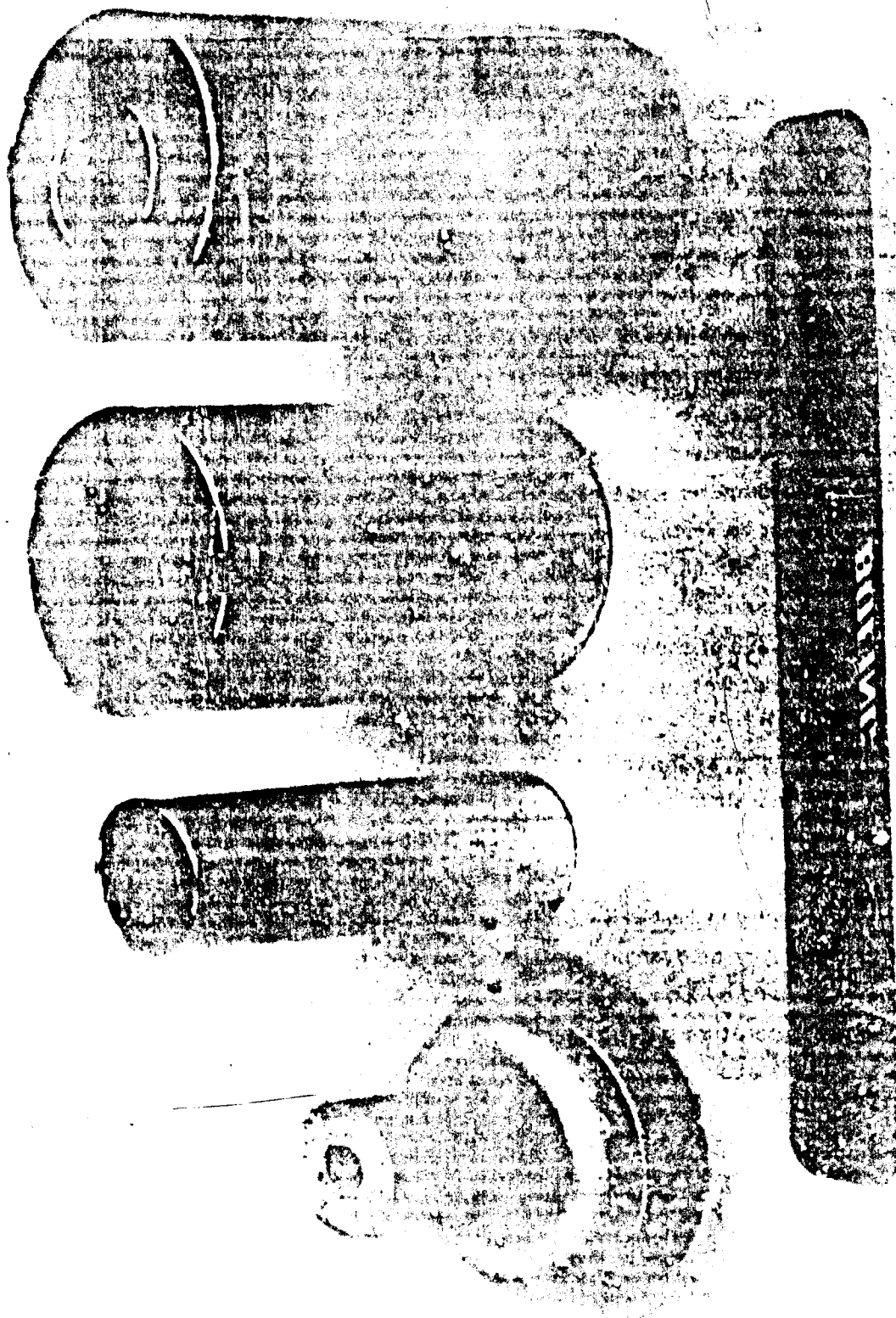


Figure 35: Concentric Cylinder Assembly

Halographic analysis has been performed by Hudgins et.al.⁴³ to determine the stress vs. strain response and Poisson's ratio measurements. Stress relief within confined walls can be studied and special formulation and construction methods can be evaluated using this procedure.

6.6 Transients. In spacecraft design, the peak voltages from pulses and transients must be calculated and the equipment must be designed to withstand the peak voltages. Transients with peak values to 160% have been recorded on a radar power supply output lines following a modulator tube arcover of a crowbar action. Systems employing pulses, short-circuiting devices, and vacuum tubes subject to arcover may be subjected to these peak pulses.

Other sources of transients are high-voltage transformers and circuits having an abundance of partial discharges with peak values exceeding 100 pC. The compact design dictates that the low-voltage and high-voltage circuits be installed in the same module with little spacing. Transients or partial discharges generated within the voids and air spaces in the high voltage winding are coupled into the low voltage circuitry by capacitive coupling through the insulation between conductors and circuit components or by common mode through the ground circuit. High speed bidirectional transient voltage suppressors may be placed on the low voltage windings of the two transformers to protect the low-voltage circuits. These suppressors may be placed in one of two locations: either at the source of the voltage transient (at the transformers) or near the parts they are designed to protect.

6.6.1 Identifying Conducted EMI Sources. There are two primary areas within the electronic circuits where EMI is generated. These are the AC-to-DC conversion package and switching circuitry (assuming switch mode conversion is implemented) and the transients and partial discharges within the high-voltage circuitry. Some energy may be conducted to the low-voltage control/sensing integrated circuits(IC's) via the control lines. This can cause controlling errors, or, if severe enough, overstress the IC and resulting in permanent damage. EMI may also be conducted from the source or load back through the supply and then to the regulatory IC. This case must not be overlooked. Many times, especially in high-voltage, momentary shorts occur across output terminals which have fast rise and fall times (less than 1 microsecond).

6.6.2 Limiting Conducted EMI Effects. Once the sources of conducted EMI are known by tests and evaluation, calculations may be made as to their magnitude and location within the frequency-domain. This is accomplished by knowing maximum voltage and current levels used within the EMI sources, semiconductor rise and fall times, and power supply input and switching frequencies. Many papers concerning these methods have been written detailing these calculations⁴⁴.

After EMI levels are established, techniques may be used to filter or suppress the EMI to acceptable levels which will not damage the low voltage controlling IC's⁴⁵. This must be done in such a manner that normal operation is unaffected.

6.6.3 Identifying Radiate EMI Sources. Radiated EMI comes in two forms, that is, it is produced from both E and H fields. E field radiation varies directly with the voltage of its source. H field radiation is dependent upon current, number of turns, and the loop area of the current.

As in conducted EMI, radiated EMI is produced wherever switching and commutating circuitry are located. The interference sources extend into magnetic devices (transformers, inductors) which emit magnetic fields. Also, component leads, wiring, and printed circuit strips can all transmit EMI.

6.6.4 Limiting Radiated EMI Effects. In general, radiated EMI can cause erroneous compensation in power supply outputs due to bias level shifts in IC operational amplifiers and associated circuitry. Metallic shielding (Faraday shielding) of rectifiers, magnetics, and switching circuitry, reduces the EMI transmission⁴⁶. Controlling (slowing) transistor and diode rise and fall times (snubbers) can reduce EMI generation⁴⁷. However, this can increase power dissipation within the supply, reducing efficiency and increasing heat sink requirements.

6.7 Grounding and Bonding. There are plans for spacecraft that will have power requirements of a few watts to 2.5 megawatts. The bonding and grounding of these units will vary considerably. Smaller spacecraft with powered loads to 5 kW will use standard, single-point grounding techniques with the solar arrays referenced to the central load module. Larger spacecraft using multiple solar array sections, capable of being transported to space via the shuttle and other transporting methods and assem-

bled in space, may have a main load center and several remote load-centers. Those spacecraft will require special bonding and grounding considerations.

6.7.1 Composite Structures. Spacecraft respond to the natural space environment by assuming a range of potentials relative to the plasma potential, depending on the plasma density, charged particle flux, and solar illumination. To equalize the potentials, it is necessary to maintain continuous electrical paths throughout the structure. Present spacecraft with composite structure already use conducting coatings to alleviate the spacecraft charging problem with reasonable success. The use of a composite structure will change the nature of the spacecraft ground and complicate grounding procedures. Concerns such as electrical continuity through the structure will become more important whenever a composite joint is encountered. However, initial results with composite spacecraft have indicated that composite joint designs are workable when properly grounded and bonded. There is still a potential corrosion problem. The design criteria shown in Table 10 have been established for composite/metallic joints in spacecraft (references 31 and 48).

6.7.2 Composite Joints. Conduction between two composite members may be provided by a screen joint, an adhesive bond using a conductive metal-filled epoxy, or by metal fasteners such as rivets or bolts⁴⁹. Six joint configurations are shown in Figures 36 through 41 and an assessment of them is shown in Table 11. Metal connectors (figure 40) are recommended for large space structures. Although the initial cost is high, the ease of fabrication in space may make this system the most cost effective. The second best method is a screen technique shown in Figure 37. The metal splice (Figure 41), mechanical fastener (Figure 39), and metal filled epoxy (Figure 38) need much improvement or research and development before they can be considered as applicable for assembly in space.

Static Drain. To support the use of graphite-epoxy composite structures in space, joints must be developed to provide electrical conduction between composite structural members for static drain and for a fault current return path. The static drain path is necessary because the effect of vehicle charging can be detrimental where the conducting sections of the vehicle are not bonded together. For example, consider a vehicle that is charged triboelectrically on the forward surfaces and discharged through corona or the plasma from the skirt at the edges. If the forward section is not electrically connected to the aft section, charge acquired on the forward section

TABLE 10

RECOMMENDATIONS IN DESIGNS WHERE GRAPHITE/EPOXY IS
COUPLED WITH OTHER MATERIALS, FOLLOW THE RULES BELOW:

METAL GROUPING

| I | II | III | IV |
|---|---|--|--|
| MAGNESIUM AND MAGNESIUM ALLOYS | ALUMINUM ALLOYS, CADMIUM AND ZINC PLATE | LEAD, TIN BARE IRON AND CARBON OR LOW ALLOY STEELS | CRES, NICKEL, AND COBALT BASED ALLOYS, TITANIUM, COPPER, BRASS, CHROME |

- DO NOT COUPLE GROUP I, II, OR III METALS DIRECTLY TO GRAPHITE/EPOXY.
- WHEN GROUP I, II, OR III METALS ARE WITHIN 3 INCHES OF GRAPHITE/EPOXY AND CONNECTED BY AN ELECTRICALLY CONDUCTIVE PATH THROUGH OTHER STRUCTURES, ISOLATE* THE GRAPHITE/EPOXY SURFACES AND EDGES.
- TITANIUM, CRES (A286 OR 300 SERIES STAINLESS STEEL), NICKEL, AND COBALT-BASED ALLOYS MAY BE COUPLED TO GRAPHITE/EPOXY STRUCTURES. WHEN OTHER GROUP IV METALS ARE COUPLED, ISOLATE* THE GRAPHITE/EPOXY SURFACES AND EDGES.
- * ISOLATION SYSTEM:
 - ONE LAYER OF TEDLAR; OR TYPE 120 GLASS FABRIC WITH A COMPATIBLE RESIN; OR FINISH.

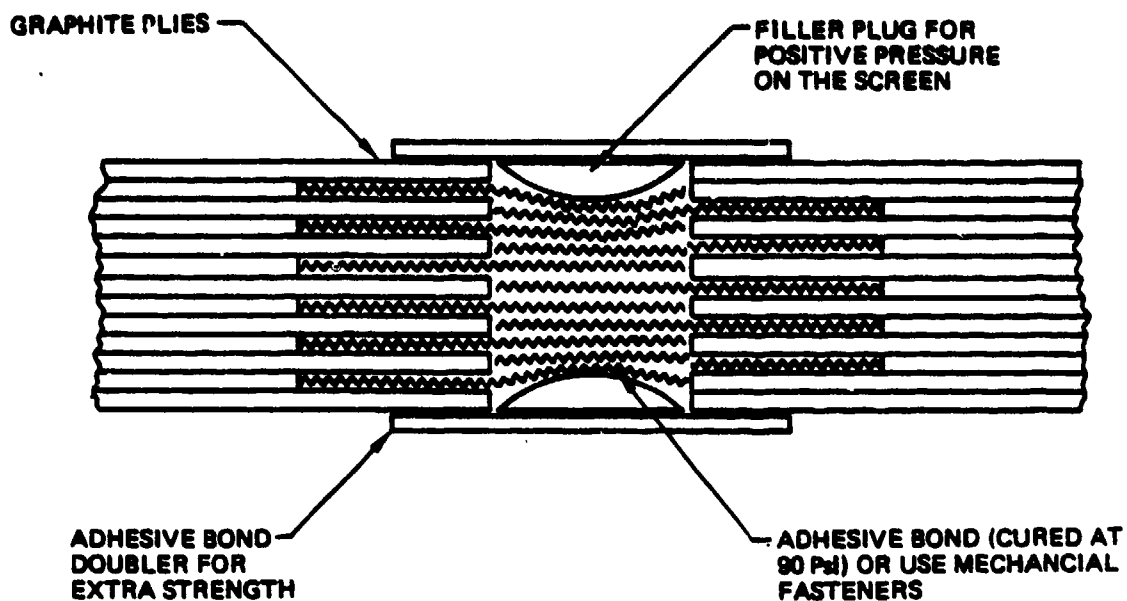


Figure 36: Multiple Screen Interleaved Lap Joint

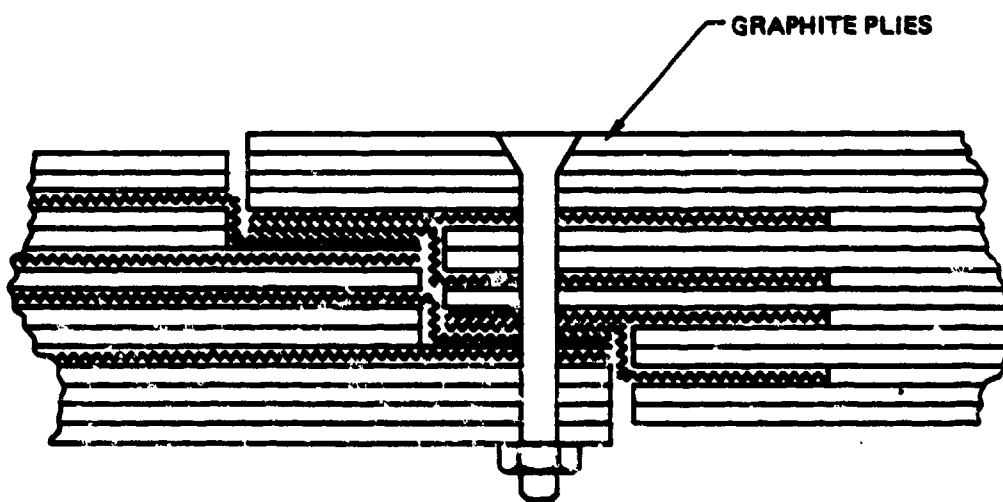


Figure 37: Multiple Exposed Screen, Mechanically Fastened Stepped Lap Joint

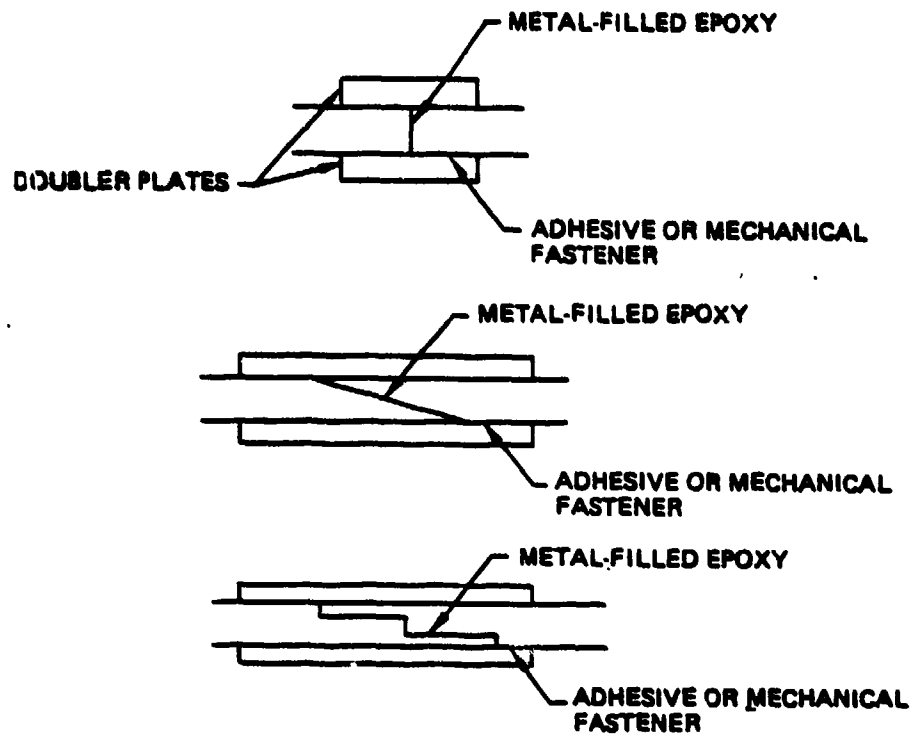


Figure 38: Butt, Scarf, and Stepped-Lap Joints

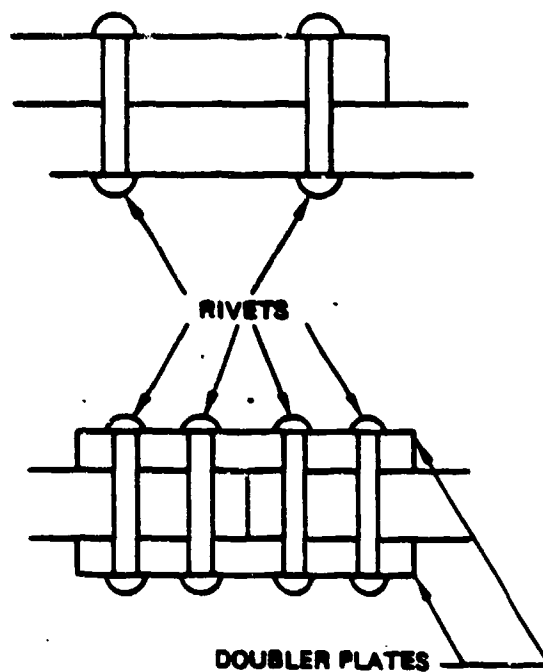


Figure 39: Mechanically Fastened Joints

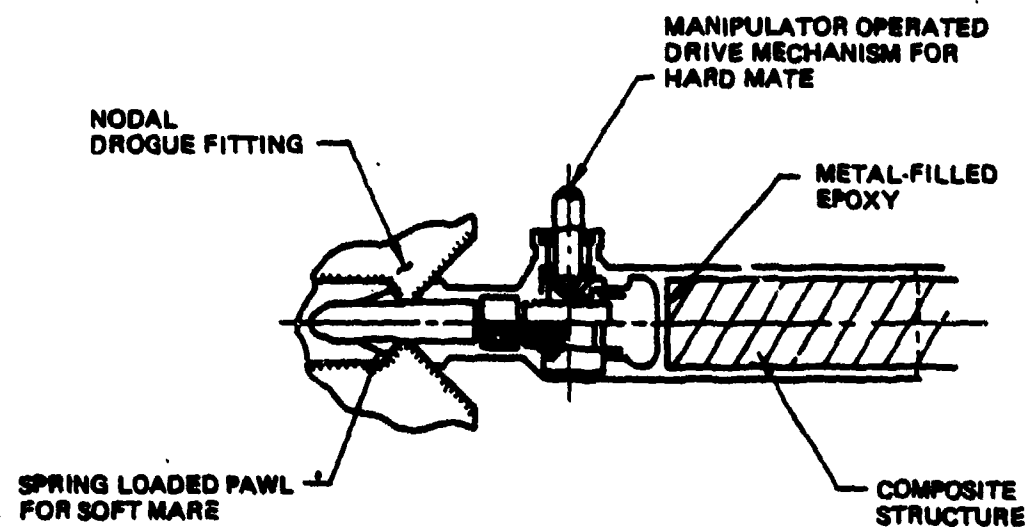


Figure 40: Metal Connector

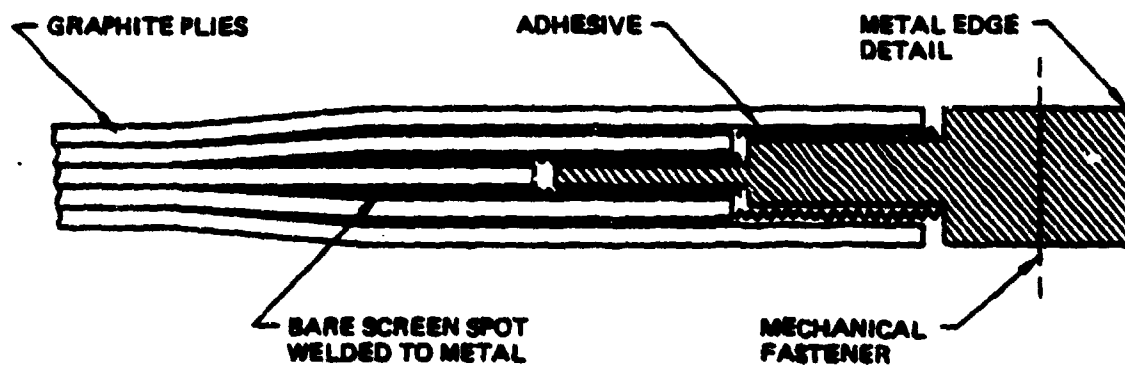


Figure 41: Center Screen Stepped Lap Composite to Metal Joint

TABLE 11: BONDING/GROUNDING CONCEPT ASSESSMENT

| <u>Joint</u> | <u>Advantages</u> | <u>Disadvantages</u> |
|-----------------------------------|---|--|
| Screen (Fig. 39) | .Good electrical conductor | .Difficult to fabricate in space .Requires individual component layup .Requires doublers for mechanical strength |
| Screen (Fig. 40) | .Good Electrical conduction .Inherent mechanical strength .Can be fabricated in space with pre-launch preparation | .Requires individual component layup |
| Metal-filled Epoxy (Fig. 41) | .Good Electrical conduction .Can be used on cut ends of continuously formed members | .Difficult to fabricate in space .Requires doublers for mechanical strength |
| Mechanical Fasteners (Fig. 42) | .Inherent mechanical strength .Can be fabricated in space .Components may be joined at positions other than ends .Allows use of continuously formed members | .Poor electrical conduction |
| Metal Connectors (Fig. 43) | .Good electrical conduction .Inherent mechanical strength for truss structures .Easily fabricated in space with pre-launch preparation .Can be used on cut ends of continuously formed members | .Requires expensive and heavy connectors .Limited to truss structures |
| Metal Splice (Fig. 44) | .Good electrical conduction .Inherent mechanical strength strength for joining panels | .Cannot be fabricated in space .Requires individual component layup |

cannot flow to the aft section unless the potential difference between the sections becomes large enough for a spark discharge to occur. These spark discharges can be quite energetic, since the capacitance between the sections may be several thousand picofarads and the sparkover voltage may be several kilovolts. Furthermore, the spark discharge will seek the easiest electrical path between the sections. If there is some electrical wiring routed across this gap, it is possible that the spark will travel through a shorter gap from the section to the wiring, through the wiring, and then through another short spark gap to the aft section. This, of course, would put a tremendous noise pulse on any data line. Also, there is the possibility that these spark discharges could fire electro-explosive devices.

6.8 Multipactor Or RF Breakdown. At frequencies greater than 10^6 Hz, a new phenomenon begins to be important in breakdown behavior. This phenomenon, called multipactor, is associated with charged particle resonance. The effect of multipacting on the Paschen curve at pressures below 10^{-3} Pa is illustrated in Figure 42. Most of the communications between spacecraft and ground stations takes place at frequencies in the 20 to 30,000 MHz range.

The design and fabrication practices for this frequency range are different than those for DC. These differences are due mainly to multipactor.

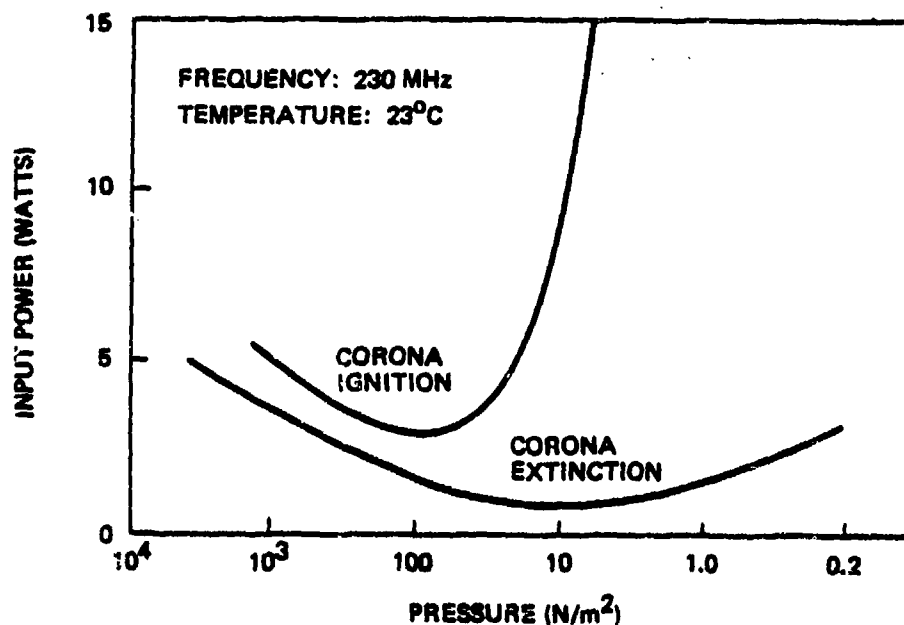


Figure 42: Quadriplexer Breakdown Characteristics

The effect does not occur until the mean free path is larger than the gap, i.e., below the Paschen minimum. The multipactor discharge is basically a resonance phenomenon, as shown in Figure 43. The electric field of the RF oscillations accelerates the electrons in the gap. When the electron strikes the electrode it can produce secondary emission. If the frequency and phase are just right, the secondary electrons will be accelerated toward the other electrode and the process will repeat itself. Multipactor is thus a resonance effect. The upper formula in Figure 43 gives the relation between gap size (d), amplitude of the electric field (e) and frequency ($\omega=2\pi f$). The breakdown voltage, V , is also given. This voltage is over-estimated by the factor due to the simple theory used⁵⁰.

The transmitting system is affected in many different ways by multipactor. There is some RF power lost in exciting the electrons. When sufficient gas molecules are present ionization can occur, which can cause corona with the resultant breakdown between plates. The impact of the electrons on the surface can cause heating and

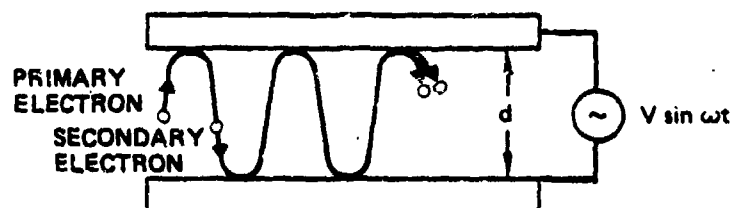


Figure 43: Multipactor Discharge: Electron Resonance in an RF Field With Discharge Sustained by Secondary Emission

outgassing. The load on the RF source is reactive which causes detuning of the output circuits. An increase in the harmonic output from the transmitter is caused by this non-linear load. Also, noise is generated by multipactor effects which can interfere with nearby receiving equipment. All of these effects will probably not happen together. The antenna may be perfectly useful as a radiator after breakdown, but its efficiency with respect to total radiated power will be drastically reduced as shown in Figure 44 for a RF coupler with intermittent multipactor.

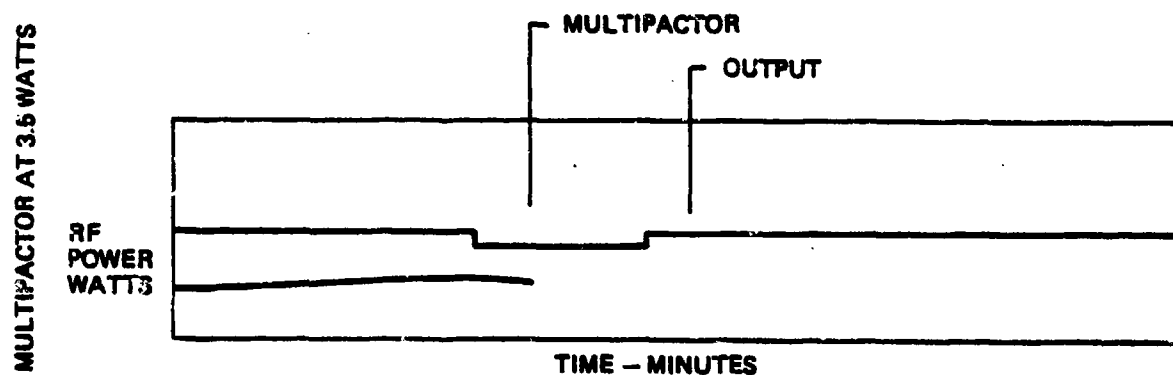


Figure 44: Power Decrease Due To Multipactor

Someone not familiar with multipactor effects could interpret these symptoms as indicative of some other cause. This may have happened in the loss of the S-band subcarriers 18 in Apollo 7. The report states: "The failure was characterized by:

- a. Drop in the ground-received PM signal strength
- b. Loss of PM subcarriers
- c. Lower than expected transponder-received signal strength.

No other abnormalities were detected. The only components within the S-band system which could have failed and caused all these symptoms are the panel switch for selecting the primary or secondary transponder and the wiring which controls this function. The switch was X-rayed and functionally tested postflight with no abnormalities noted. The transponder was tested in the command module and on the bench, including vibration and temperature acceptance testing, and the results were all negative.

When the select switch is changed from one transponder to the other, a momentary hesitation in the OFF position is required to allow latching relays to reset. Switching without this hesitation can cause both transponders to be ON and will create all the symptoms of the failure. The transponder select switch, directly above the antenna select switch, may have been inadvertently thrown during one of the frequent antenna switchings, and both transponders may have been activated. Although the crew member on duty cannot remember inadvertently throwing the wrong switch, he does not discount the possibility.

No further action is required, and this anomaly is closed. Symptoms (a) and (c) could easily have been due to multipacting. Symptom (b) may be caused by other phenomena. The thing to note is that multipactor was not even considered as a possible failure mode. Also, the suspected elements showed no abnormalities even when X-rayed because multipactor, like a glow discharge, has little heating or burning involved. This possible misinterpretation of an anomaly just points out the need for greater assimilation of knowledge about electrical discharge.

Electrode spacing, frequency, and the applied voltage must all be considered when considering design parameters for the elimination of multipactor. Figure 45 displays one set of relationships among these parameters. The peak voltage is the ordinate while the product of frequency and electrode separation is the abscissa. The scale at

the top is electrode separation divided by wavelength, both in the same units. The three regions indicated by $1/2$, $3/2$, and $5/2$ are where multipacting can occur. The electrons that correspond to the region marked $1/2$ have a transit time of $t/2$, where $t = 1/F$. Those in region $3/2$ have a transit time of $3t/2$, etc. Unfortunately, a general plot of this sort cannot be relied on for design data, because the values depend on factors such as number of electrodes, electrode material, surface conditioning of electrodes, and geometry (e.g., parallel plate, coaxial, etc.). The designer is forced to test each design at various stages of fabrication to be certain that multipacting is not occurring.

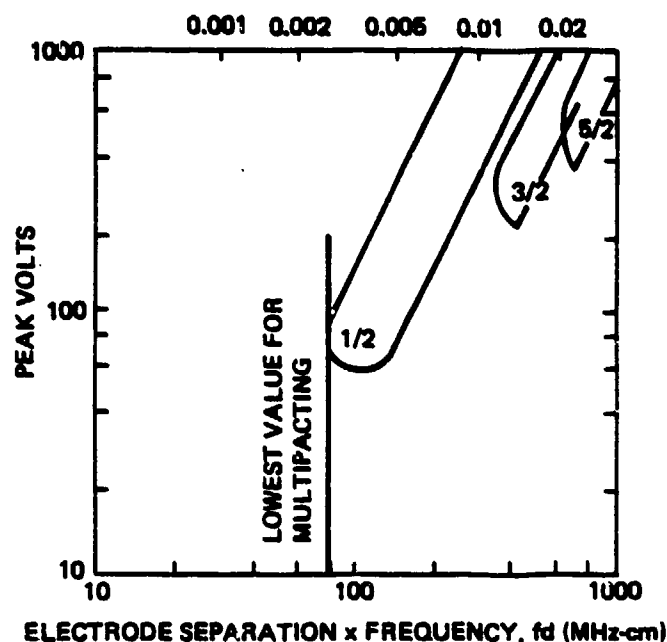


Figure 45: Possible Regions of Multipacting Between Parallel Plates

Some design features that have been known to help are³:

- The insertion of foamed dielectric material between the electrodes
- Treatment of electrode surfaces to change their secondary emission characteristics
- The application of dc bias to the electrodes to suppress secondary electron emission
- Pressurization of the space between the electrodes
- The selection of electrical and mechanical dimensions to reduce the likelihood of multipaction

Before anything can be done to prevent multipaction it has to be detected. There are three prevalent methods of doing this. One is the observation of the faint glow which results from ionization of the gas. However, encapsulation would eliminate the use of this method. Phosphorescent materials placed near the multipactor area will glow under bombardment of the electrons. This could affect the conditions existing in the area. The most sensitive and satisfactory detector consists of a collector electrode in the suspected area. A dc current will flow to the electrode if a small dc voltage is applied and a discharge is present.

6.9 The Malter Effect or clumping. The "clumping" mechanism occurs when the electric field is large enough to remove a charged particle of material from one electrode and then accelerate it to the opposite electrode. The impact releases enough energy to produce localized heating which creates a vapor cloud. This is usually followed by voltage breakdown.

This type of field emission is caused by an impure cathode. The surface charges can be caused by photons from a preceding discharge or from an outside source. This lowers the breakdown voltage for values of Pd below the Paschen minimum.

6.10 Spacecraft Charging. There are two important charging mechanisms: triboelectric charging and plasma charging.

6.10.1 Triboelectric Charging. Triboelectric charging occurs whenever two dissimilar materials are placed in contact with each other and then separated. One material pulls electrons from the other leaving the first (the one with excess electrons) with a negative charge and the second with a positive charge. An example of this occurs when an aircraft flies through precipitation containing ice crystals; the ice crystals lose electrons to the aircraft. Thus, the aircraft ends up with a negative potential due to the accumulation of negative charge. The potential of an aircraft flying in precipitation will rise until the corona threshold potential is exceeded. At this point corona discharge will begin and the corona discharge current will be equal to the charging current. It is a good approximation to regard the triboelectric charging mechanism as equivalent to a constant current source.

Rocket Motor And Jet Engine Charging. Before any conclusive measurements on rockets it was felt that the rocket engine could have one of three effects on the electrostatic potential of the spacecraft:

- a. If the engines were capable of charging the vehicle to potentials above the vehicle threshold potential, then corona discharges and electromagnetic interference would accompany each launch.
- b. If the conductivity of the exhaust was sufficient to limit the vehicle to some value below the corona threshold potential, then there would be no corona discharges or associated electromagnetic interference due to engine charging. Other charging sources could, however, still cause corona discharge.
- c. If the ionized rocket exhaust was so effective a discharger that the vehicle potential was held below the corona threshold even when other charging sources were present, then the rocket engine would tend to alleviate, rather than aggravate, the vehicle charging problem.

From measurements made on Titan III C rockets it is known that the rocket exhaust is a good discharger⁵¹. Jet engines correspond to the second possibility.

In order for a discharge to take place into the air it is necessary that the field be sufficiently high that an electron (on the average) can acquire ionizing energy between collisions. In other words, the energy of the electron depends on the electric field and the mean free path. This means that the discharge will take place at places with the smallest radii of curvature, i.e., the small radius concentrates the field lines. As a rule, the discharge will occur from burrs and imperfections (unless dischargers are present) which exist on the extremities of the aircraft.

For the discharge to begin there has to be at least one free electron (which was probably produced by cosmic rays). The action of the electrostatically produced field moves this electron from the point and causes it to collide with air molecules which become ionized. This produces additional electrons which, in turn, are accelerated by the field. This electron avalanche continues to propagate and grow until it reaches a region at some distance from the point where the field is too low to permit ionization by collision and where the electrons are slowed sufficiently that they attach to oxygen

molecules to produce δ_2 ions. The δ_2 ions are much less mobile than the electrons and they may be considered to be stationary as far as the discharge process are concerned. The relatively stationary cloud of δ_2 ions tends to reduce the field between itself and the point. This reduces the distance to which the next avalanche can propagate. On the other hand, the cloud of positive ions left behind by the movement of the electrons tends to increase the field between itself and the point. Thus, avalanches are initiated more readily in this region. Many of the free electrons necessary for avalanche formation are probably supplied by photoemission from the negative point. The discharge continues as a series of successive avalanches.

Each avalanche propagates a shorter distance than the last as the inner limit of the cloud of δ_2 ions approaches the discharge point. Meanwhile, the positive ions are being drawn into the point. Finally, the negative space charge reduces the field near the discharge point to such an extent that ionization by collision is no longer possible. When this occurs the discharge is choked off. At sea-level pressures this process only takes about 0.2 μ secs. Under the action of the wind and the applied electric field the ions are gradually swept away from the discharge point. The whole process then repeats itself.

The potential at which discharge occurs goes down with an increase in altitude. The reason for this behavior is exactly the same as for the case of two electrodes, i.e., the mean free path increases with decreasing density.

The discharges occur as a series of discrete impulses of short duration and rapid rise time. They therefore produce RF noise over a broad spectrum. This interference may disable radio receiving systems and, in some cases, may induce spurious pulses in the electronic systems controlling stage sequencing and vehicle guidance.

Discharge Due To Improper Bonding. Another place where the effect of vehicle charging can be detrimental is where the conducting sections of the vehicle are not bonded together, for example, if a rocket vehicle is charged triboelectrically on the forward surfaces and discharged through corona from the skirt at the aft end. If the forward section is not electrically connected to the aft section, charge acquired on the forward section cannot flow to the aft section unless the potential difference between the sections becomes large enough for a spark discharge to occur. The electrical isolation could occur as a result of improper electrical bonding at the interface of two sections.

These spark discharges can be quite energetic, since the capacitance between the sections may be several thousand picofarads and the sparkover voltage may be several kilovolts. Furthermore, the spark discharge will seek the easiest electrical path between the sections. If there is some electrical wiring routed across this gap, it is possible that the spark will travel through a shorter gap from the front section to the wiring, through the wiring, and then through another short spark gap to the aft section. This, of course, would put a tremendous noise pulse on any data line. Also, there is the possibility that these spark discharges could fire electro-explosive devices. Proper bonding between sections of the spacecraft is mandatory.

Streamers From Insulators. Vehicle charging can also be detrimental when the vehicle skin is composed of dielectric or of dielectric-coated sections which can become charged triboelectrically from passage through ice crystals or other particulate matter. In contrast with the metal skin, the charge cannot flow away from the point where it is deposited. Charge thus accumulates on the surface until the electric field along the surface is large enough to support a streamer discharge over the dielectric surface to a metal structure nearby. However, if the dielectric strength of the insulator is exceeded before the streamer occurs, then the charge is relieved by a spark discharge that punctures the dielectric and travels to an underlying conductor. Streamer discharges, like spark discharges, seek the easiest path to the vehicle structure.

Windshields. Since windshields are made of an insulating material the same sort of effects occur with them as with other dielectric materials. Streamer discharge from windshields is a source of RF noise.

Staging Effects. It had been conjectured that electrostatic discharges could occur between stages as they separated. In fact, it was shown that the separation of two dissimilar objects could cause substantial voltage difference between the bodies⁵². However, experiments by Vance and Nanevich⁵³ have shown that the two parts of the staging vehicle will be electrically connected through the conductive exhaust plume as long as the motor exhaust plume lays on the expended stage. It does not seem conceivable that significant differences in potential (i.e., more than a few tens of volts) can be developed between separating sections during a staging event in which the upstage motor is ignited at or before the time of stage separation.

Coating Insulators and Windshields. It has been found that a high-resistance conductive coating over the dielectric surface is quite effective in eliminating streamer noise. The conductive coating drains away the charge as rapidly as it arrives and prevents the electrostatic potential build-up which produces the streamer discharges.

The coatings⁵⁴ used for non-transparent dielectrics are usually opaque and have a surface conductivity on the order of a (Megohm)⁻¹.

Most windshields are made of one of either glass or acrylic plastics. Glass has a lower surface resistance, 10^{12} ohms, than the acrylics, (10^{16} ohms). This is attributed to the somewhat open silica network in glass which allows hydration. It has been shown that a surface resistance of 10^8 ohms is probably sufficient to bleed off accumulating charge.

The principal coating presently used for glass outer panels is stannous-oxide, which can be fused into the glass exterior surface to sufficient depth that erosion should not seriously reduce the conductivity of the external surface coating during the life of the windshield.

Changing the Coupling Between The Antenna And The Dischargers. When it is only the communication link that is of importance a great deal of noise suppression can occur through the proper choice of the antenna and its placement. The equivalent noise field of a loop antenna can be reduced by mounting it near the end of an airplane member⁵⁵. The correct placement can reduce the noise factor by 25 db. At least 25 db of noise suppression can also be obtained with dipole antennas if two of them are correctly placed and balanced.

Discharge Effects On Electro-Explosive Devices. Electro-explosive devices in the rocket can be set off by corona discharges. Unbonded sections present a real danger because of the intensity of the spark and because the spark may discharge through an electro-explosive device. Streamer discharges can also be energetic enough to ignite one of these devices.

6.10.2 Power Loss By Leakage Through Plasma. The space between 300 km altitude and the orbits of geosynchronous satellites contains neutral atoms, free electrons, positive ions, and high-energy charged particles. These high-energy particles, although damaging to solar cells and optical surfaces, are not numerous enough to carry a significant current. The free electrons, generated when ultraviolet photons ionize neutral atoms, have energies of around one to two electron volts. This energy is dissipated in reactions with neutral atoms and ions, increasing the temperature of the medium to the region of 500° to 2000°K. The temperature of an electron is related to its energy by Boltzmann's constant, 8.6171×10^{-5} eV per °K.

An electrically neutral gas containing free electrons and ions in equal numbers is called a plasma. A positively charged spherical electrode, say one cm in diameter, will collect electrons when inserted into a plasma. The volume in which electrons are influenced by the electrode, called a sheath, is much larger than the sphere. Some of the electrons will orbit around the electrode and escape back out of the sheath. Current collection is then said to be orbit-limited and is affected in a complex manner by the radius of the electrode, the voltage of the electrode, and the temperature and density of the free electrons.

The high-voltage solar-cell array for a high power satellite looks more like a sheet electrode than like a spherical probe. K. L. Kennerud has developed a method of analyzing the leakage current from such arrays (Reference 56) based on fundamental equations developed by I. Langmuir (Reference 57). Kennerud's technique converts the linear array into a sphere having the same area, and then he calculates the radius of the electron sheath surrounding the array. His experiments with small positively charged solar-cell panels correlated well with his predictions. With a negatively charged panel which collected ions, his experimental measurements did not correlate well with theoretical predictions, perhaps because the ion sheath extended to the chamber walls.

Using Langmuir's equations, it can be determined that at 500 km the electron sheath extends to a few meters above the plane of the solar cells, in the range of electron concentrations, electron temperatures, and array voltages of interest. The calculation of leakage current then simplifies into analyzing the rate at which electrons drift into an electron sheath having essentially the same area as the solar array. The electron current J_r (A/cm²) is simply:

$$J_r = \frac{N_e \sqrt{E_e}}{37 \times 10^{11}}$$

where N_e = electron density, in electrons per cm^3

E_e = electron energy in eV

A flow of electrons from the plasma to a high power satellite must be matched to an equivalent flow of electrons out of the satellite. Otherwise the satellite will become negatively charged with respect to the plasma, and will cease attracting electrons. This flow of electrons away from the satellite is provided during orbit transfer by electron emitters which are installed for neutralizing the ions emitted by the thrusters. In geosynchronous orbit, where the satellite would be generating power, the electric thrusters may not be in operation continuously. Furthermore, in geosynchronous orbit the electron density is only about 100 per cm^3 , so the power lost through plasma leakage, even at several kilovolts, would be trivial.

A negatively charged solar array would attract ions rather than electrons. However, ions are less mobile than electrons, and the ion current would be much smaller than the electron current observed with a positively charge solar array. Thus, the positively charged array is the worst case.

Irving Langmuir, in Reference 57, provides the following equation for calculating electron current from plasma to a positive electrode:

$$I = 4 r_0^2 J_r \text{ if } r_0 < p$$

where

$$r_0 = \sqrt{\frac{D}{J_r \alpha^2}} = \text{Space-Charge Sheath Radius (cm)}$$

$$D = 2.336 \times 10^{-6} (\beta V_p)^{3/2}$$

α^2 = a function of r_0/a which is calculated as shown below.

$$J_r = \text{random current density of plasma electrons (amps/cm}^2\text{)} = \frac{N_e \sqrt{E_e}}{37 \times 10^{11}}$$

$$p = a \sqrt{1 + \frac{\beta V_p}{E_e}} = \text{impact parameter (cm)}$$

a = radius of sphere having same area as array

β = fraction of sphere surface area uncovered

V_p = potential applied to array, volts

E_e = average energy of electrons

N_e = electron density (electrons per cm^3)

I = electron current collected by the sphere (amperes)

Langmuir's table relating $\frac{Y_0}{a}$ to α is not applicable to the large electrode areas involved in high power satellites. The value of α was determined by iteration of the equation,

$$\alpha = \gamma - 0.3\gamma^2 + 0.075\gamma^3 - 0.0143182\gamma^4 + 0.0021609\gamma^5 - 0.00026791\gamma^6$$

where

$$\gamma = \log_e \frac{r_0}{a}$$

The analysis technique developed by K. L. Kennerud wraps the solar array area (A_a) around a sphere, which then has radius a .

The above equations can be used to estimate the leakage current for positive ions or electrons.

The calculations for leakage current shown in Table 12 were based on the Figure 46 electron densities and electron temperatures from Reference 58.

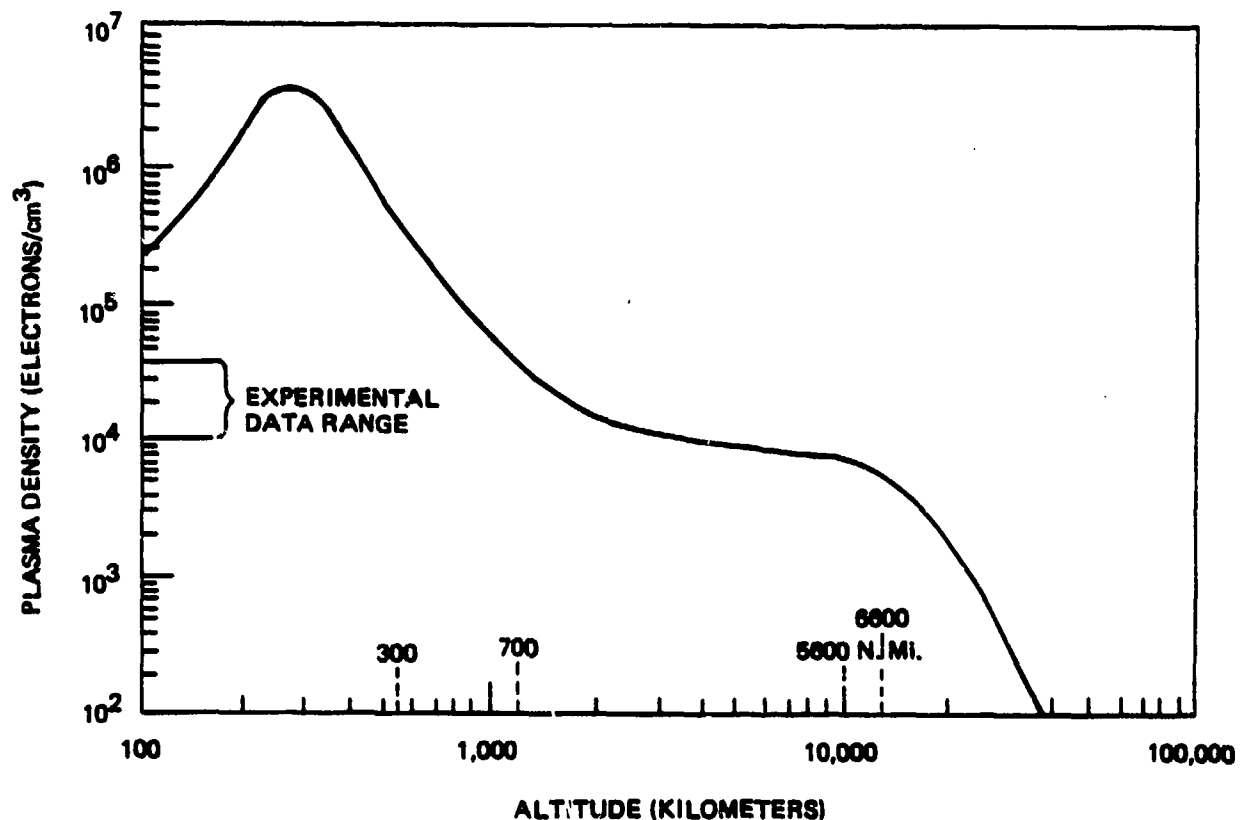


Figure 46: Plasma Density Versus Altitude

TABLE 12. LEAKAGE CURRENT FROM POSITIVELY CHARGED
SOLAR ARRAY

| Array Altitude, Km | Electron Density, N_e Electrons/cm ³ | Electron Temperature °K | Leakage Current | | Power Loss, Percent of Generated |
|--------------------------|---|-------------------------------|--------------------|-------------------------------|--|
| | | | nA/cm ² | Amperes per 1500 V String* | |
| 500 | 6×10^5 | 3,000 | 824.5 | 0.8494 | 7.72 |
| 700 | 2×10^5 | 3,000 | 274.8 | 0.2831 | 2.57 |
| 1,000 | 7×10^4 | 3,000 | 96.19 | 0.0990 | 0.90 |
| 2,000 | 2×10^4 | 3,200 | 28.38 | 0.0292 | 0.265 |
| 5,000 | 1×10^4 | 4,400 | 16.64 | 0.0171 | 0.156 |
| 10,000 | 8×10^3 | 5,400 | 14.75 | 0.0152 | 0.138 |
| 20,000 | 2×10^3 | 9,000 | 4.76 | 0.0049 | 0.044 |
| 30,000 | 1×10^2 | 13,600 | 0.29 | 0.0003 | 0 |

* The string is 0.404 m by 255 m, with an area of 133.02 m²

Solar Array Power Loss. Experiments by Fralick⁵⁹ show that power loss due to plasma leakage current will become significant above 100 volts, and above 200 volts arcing was observed with both positive and negative biased solar array samples. Leakage current as a function of voltage is shown for both positive and negative biased solar arrays in Figure 47. The solar array area was 1.37 m^2 and the interconnect area was approximately 0.14 m^2 . Although the current is much smaller for the negatively charged solar array, arcing does occur at voltages above 200 volts dc. These experiments and others by Kennerud indicated that solar arrays in low Earth orbit may experience significant power leakage loss if positively biased, or severe arcing if negatively biased.

Localized arcs and glow discharges through the plasma could certainly be tolerated for a short period of time. The problem is the long term effect of surface shorting between conductors and the noise generated and radiated/conducted to the spacecraft through the power lines and atmosphere. Noise peaks of 50 volts and 250 volts were recorded for positive and negative biased solar arrays as shown in Figure 48.

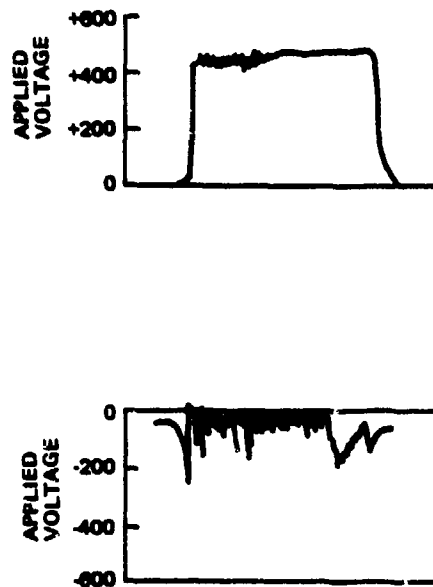


Figure 48: Noise Voltage Profiles for the Solar Array Panels

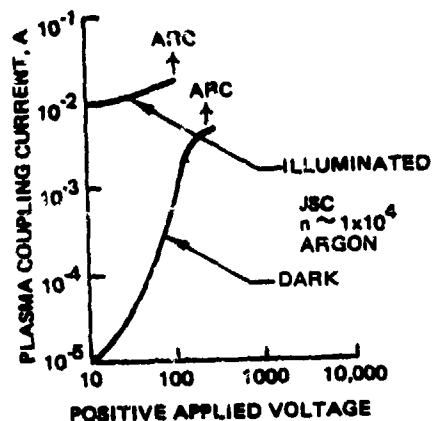


Figure 47A: Plasma Coupling Current as a Function of Positive Applied Voltage for the Nine-Panel Array in Dark and Fully Illuminated.

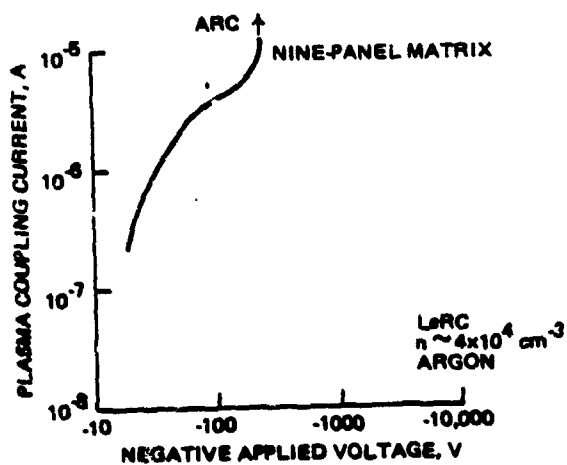


Figure 47B: Plasma Coupling Current as a Function of Negative Applied Voltage for the Fully Illuminated Solar Array Samples.

The significance of insulating or not insulating the solar array interconnects should not be overlooked. An insulated solar array can withstand a much greater bias voltage, positive or negative, than a non-insulated array. The problem is obtaining a 100% insulated array. Should there appear a very small pinhole greater than 0.01 mm dia, any place on an interconnect plasma current will be emitted from that hole. The current through the pinhole will be very large compared to the insulated areas surrounding the pinhole. This may cause conductor heating and loss of an interconnect. In addition, the heating of the pinhole area will heat the insulation, deteriorating the insulation band and integrity, causing the pinhole to grow in size and effectively become a bare conductor.

6.11 Design Considerations. Some parts and components deserve special attention in their preparation packaging and installation in a high voltage circuit. Simple precautions in the selection of parts and packaging may result in a significant difference in operating life.

6.11.1 Connectors. Connectors with small gas-filled spaces on the ends of the pins or around the pins are candidates for partial discharges and eventual breakdown (Figure 29). Venting will help for low voltages but not for voltages exceeding 2000 volts, peak. Material outgassing into the wall(s) will permit partial discharges.

6.11.2 Wires. Wires interconnecting high-voltage power supplies and electronic circuits are often routed through metal wall inserts. The outer surfaces of the wires not attached to a ground plane are charged to the approximate potential of the inner conductors. This charge can dissipate or arc to the ground plane or to another lower potential wire and generate considerable noise or pulses in the circuit.

6.11.3 Feedthroughs and Terminations. High-voltage interconnections between power supplies and electronic circuits are difficult to design, assemble, and assess, especially if the interconnecting shielded wire is flexed or strained after it is attached to one or both terminals. Linear stressing can break one or all of the bonded joints between the shield, conductor, or wire insulation and the terminal and encapsulating material. Frequently, teflon-insulated high-voltage wire is used. The bond between the encapsulant and teflon is weaker than that between the metal surfaces. Any delamination at the insulation bond will form a void (air gap) and result in partial discharges and eventually voltage breakdown.

If teflon insulation is used, it is necessary to etch the teflon within a few hours before encapsulation. If the etching material is placed on the teflon at room ambient temperature more than 24 hours before encapsulation, or if the teflon is heated due to soldering, the teflon will cold-flow and ruin the usefulness of the etching. If the etching must be applied more than 24 hours before encapsulation, it is recommended that the etched surfaces be cooled to prevent cold flow.

A feedthrough should be designed with the insulator through the metal as shown in Figure 24. If metal will exist alongside the pin, voids may be trapped in the potting compound between the pin and wall. This will generate significant noise on the circuit via partial discharges.

5.11.4 Delamination. Delamination is easily detected in a transparent encapsulant. Whenever there is delamination, a silvery coloring appears along the sides or edges of the electrical part. These silvery spots are detected by transmitting polarized light through the encapsulants. Delaminations and cracks will appear in certain planes, usually emanating from the edges of components. Delamination on the surfaces of a capacitor is shown in Figure 49.

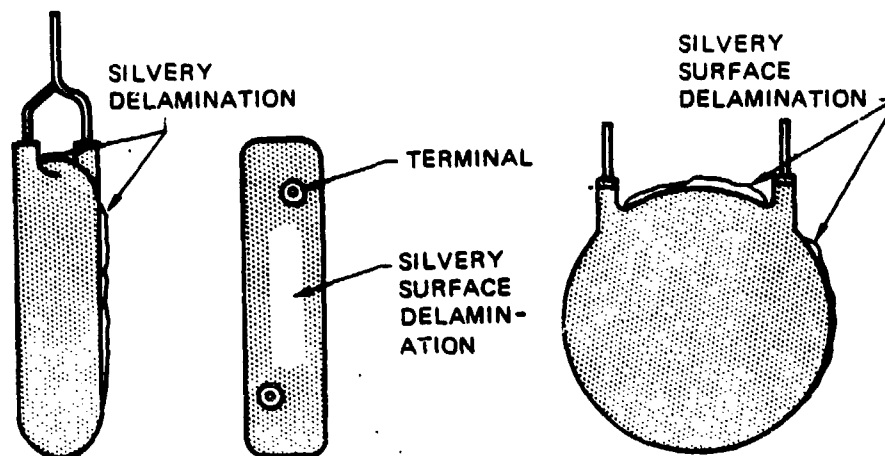


Figure 49: Delaminated High Voltage Capacitor

6.11.5 Capacitors. High-voltage ceramic capacitors for voltage multipliers should be specified to have no additives applied to the outer surface, and the capacitors should be kept in clear packages. Internally, the electrode should be deposited on the ceramic chip to prevent the inclusion of voids between a metal sheet and the ceramic. All ceramic discs should be without scratches or chips on the edges before assembly. Scratches and chips will contain voids and tracking will occur over the edges of the dielectric.

Large capacitors and inductors may be sealed in liquid or gas-filled containers. The joint between the can and insulator should not be made of solder or tin, it should be brazed. The solder in solder joints in pulse circuits tends to evaporate and deposit on the insulator surfaces, resulting in surface tracking and arcing. When soldered, an insulating conformal coat should be brushed over the solder.

6.11.6 Operating Temperature. An example of temperature increase and the effects of corona on an experiment is shown in Figure 50. A high-voltage circuit with a photo-

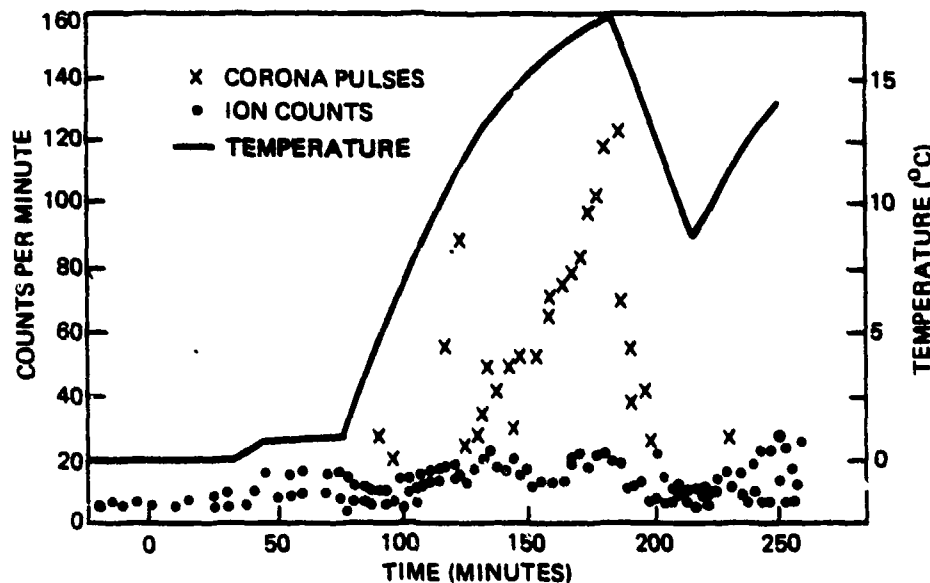


Figure 50: Partial Discharge Counts Measured in a Sensitive Photomultiplier Circuit

multiplier tube operating at 3800 volts dc, was tested at its normal operating temperature (0°) for several hours. During this period the random background noise was minimal and attributed to: spurious noises, atmospheric effects, and instrumentation. When the temperature was slowly raised to 15°C (at a rate of 15°C/hour) the noise level increased. In addition, numerous large impulses appeared at the output. These impulses were attributed to corona. When the temperature was decreased to 10°C the noise level returned to the normal threshold level.

The time lag for temperature cycling is demonstrated by the second temperature cycle starting at approximately 10°C. Note that this temperature (10°C) is almost at the critical point. Thus, the corona pulses increased immediately as the temperature was raised. These corona pulses were attributed to insulation outgassing.

7.0 TEST

The purpose for testing an insulation system is to verify that it will comply to the specified electrical parameters over its service life. Testing may be used to evaluate prototype equipment to predict life, failure modes, and for determining product improvement. Testing should be non-degrading to the insulation, circuit, and parts.

7.1 Methods and Equipment. Military, NASA, and ASTM standard and specified tests are sufficient for dielectric withstanding voltage, insulation resistance, material properties, and incoming parts evaluation. Test equipment and tests procedures defined in paragraph 5, "High Voltage Design Guide: Aircraft", of this report are adequate for spacecraft high voltage parts, modules, and systems. Two tests unique to spacecraft equipment partial discharge and pulse, are discussed in this paragraph.

7.1.1 Pulse Tests. Pulse tests, like dielectric withstanding voltage (DW) tests, can be destructive and must be carefully planned and executed. Some rules of application for pulse and dielectric withstanding voltage tests are as follows:

- a. Pulse tests peak voltage should not exceed 200% rated (peak) voltage.
- b. Pulse and DWV tests should be limited to incoming inspection, component acceptance and system/subsystem acceptance. Overtesting overstresses the dielectrics and some critical electric parts.
- c. Tests beyond the group outlined in (b) should be at reduced value 80-85% original value.
- d. The DWV and pulse tests must be magnitude limited tests. That is the magnitude must be limited to coincide with the dielectric stress within the test article insulation system. For instance, capacitors in pulse-forming networks are already rated near the electrical stress limit of the insulation system.

Pulse test techniques and equipment are described in Volumes 1, 2, and 4 of this report.

Components and equipment recommended for pulse tests are cable assemblies, capacitors, and inductors used in high-voltage, high-power equipment. Pulse tests are

not recommended for delicate experiments or very low power equipment.

7.1.2 Partial Discharges. More emphasis has been placed on partial discharge testing since 1975 than any other test. Some simple rules to follow for partial discharge tests are as follows:

- a. All high voltage parts and components should be partial discharge tested.
- b. Partial discharge magnitudes should be recorded using a pulse height analyzer.
- c. Go, no-go testing must be carefully planned.
- d. Overvoltage partial discharge testing is life degrading and must be limited in total time for each component within the system.
- d. Each test must be well planned for each type of part and component. The test operator must be skilled and in direct contact with the high-voltage engineer to discuss all abnormalities, failures, and acceptable parts, components, and systems.

Partial discharge and corona measurements and equipment requirements are discussed in Volumes 1, 2, and 4 of this final report. Volumes 1 and 2 show test parameters and specifications for a few special test articles operating at voltages greater than 50kV. Equipment, tests, and test techniques are discussed in Volume 4.

When evaluating spacecraft equipment, it is not sufficient to merely measure the height of the biggest partial discharge (pC) pulse, it is more important to measure the total quantity pulses and their pulse height distribution as reported by Burnham⁶⁰. Shown in Dr. Burnham's report are the test data and analyses for several capacitors using a thermally stimulated discharge (TSD) test. This test excites partial discharges in the test sample by applying a very high stress to the dielectric under conditions which allow the charge to be stored on the dielectric surface and then to integrate it by measuring the thermally stimulated current by a TSD test. From these tests and analyses it is shown that the degradation mechanism is damage done by electrons accelerated in small voids which give rise to low level partial discharges. The cumulative effect over a long time leads to dielectric failure. This analysis could be

established for parts and components other than capacitors, provided (a) the level of partial discharges can be quantitatively measured during a life test, and (b) the actual electric field distribution in the dielectric is known.

Special Spacecraft System Test. A special partial discharge test was used to evaluate the presence of partial discharge on the space telescope. This test was used to evaluate the complex high-voltage cabling system.

Two small aluminum plates separated by a 1 mil dielectric (mylar) were bonded to the outer surface of the cables with a coaxial lead connected to the two plates. The coaxial signal lead wire was connected to the inner plate and shield to the outer plate. The plates were then tested to determine if they did not generate partial discharges before the final test. The output of the plates was fed into a high-pass filter, which passed 50 kHz and above, and a partial discharge detection system, and the signal displayed on a pulse height analyzer. With this device the initiation and extinction of partial discharges could be determined. A similar system is described in Paragraph 7.5, Volume 4, "High Voltage Design Guide: Aircraft" of this report.

8.0 PROBLEM AREAS

8.1 Production. During production of high voltage assemblies several problem areas must be overcome. Some of the more frequent happenings that go wrong in production are as follows:

- o Routed wires move during potting, causing voids and/or stress.
- o Solder connections and interface connections result in sharp points or edges, causing corona.
- o Tapes or barriers embedded in a potting compound may be a source of voids and debonding.
- o Residue from solvents may cause poor bonding, resulting in partial discharge and creepage.
- o Silicone contamination of epoxies and/or urethanes from mold release agents.
- o Contamination by mixing silicones and epoxies in the same potting facility.
- o Lack of complete outgassing of air from the wiring and parts prior to application of the potting material caused voids.
- o Not a low enough viscosity to insure complete filling of magnetic devices and densely packaged electronics, causing voids.
- o Improper use of primers and wetting agents.
- o Improper formulation of potting compounds.
- o Contaminated cooling gases and liquids.
- o Insufficient partial discharge testing.

The most predominant modes of failure in high voltage assemblies are:

- a. Partial discharges and arcing: the design engineers should assume that the most successful design and packaging will have occasional partial discharges or transients in high voltage equipment. The design must protect circuit elements through current limiting, isolation, shielding and physical arrangement, minimizing common boundaries between high-voltage and low-voltage circuits.
- b. Mechanical failures resulting in short or open circuits: mechanical failures occur frequently during cure and thermal cycling. Analysis of failed units shows a high rate of breakage and fracture at interconnecting elements to magnetics, connectors, and circuit terminals. Design and control of lead dress at terminals is critical. Selection of potting compounds with acceptable coefficients of expansion is essential, as is packaging to accommodate the predictable thermal stresses.
- c. Cleanliness is imperative to reliable high voltage assemblies: predominant in the contamination of hardware during production is the presence of oils, greases, fingerprints, and residues resulting from handling by personnel in preparation for the potting process. This results in unsatisfactory bonding of materials, and provides an initiation for tracking, creeping, and arcing.

Semi clean-room environment and discipline is required for reliable production of high voltage assemblies. Along with residue from solvents and washes, solder flux, and particles from the work bench, attention must be given to dust and particles from the surrounding air.

8.2 Pointed Surfaces And Sharp Edges. Points and sharp edge must be avoided. The following techniques may help eliminate some of the problems.

- a. Every solder connection must be round (spherical or oval) and smooth.
- b. Hardware: Nuts, screws, and bolts require special attention. Where rounded surfaces cannot be used, a semi-conducting or metal cap should be placed over the sharp-edged elements.
- c. Interface standoffs, terminals, and brackets must be designed without points and sharp edges.

- d. Cutouts, holes, slots, and indentations must be smooth and rounded, whether it be in metal or an insulating board.

8.3 Thermal Mechanical Stress Interaction. A number of costly failures in high voltage power supplies and assemblies occurs because of differential thermal expansion of the potting or encapsulation material with respect to the thermal expansion of electrical/electronic parts and packages during cure and thermal cycles. Predominant among recurring failures resulting from mechanical stress are: broken components, wire fractures at interfaces, and cracked or crazed compounds.

Engineering prototypes and preproduction units have successfully completed environmental testing and evaluation, only to encounter failures in the qualification and acceptance testing of production units, or during the early life cycle of units accepted by the customer. Analysis of units with mechanical failures include the following reasons:

- a. Stress relief of conductors at interconnecting point not properly implemented.
- b. Wires moved during potting process, thereby voiding the intended stress relief.
- c. Incorrect component mounting for proper stress relief.
- d. Inadequate soft conformal coating of sensitive components.
- e. Lack of, or insufficient quantity of filler in the potting material to control the thermal differential expansion of epoxies, urethanes, and silicones.
- f. Cracking at low temperatures because the operating temperature and test temperature of the circuit were above glass transition temperature of the potting compound.
- g. Debonding.

8.4 Air Gaps And Voids In Insulation. Partial discharge, corona, and arcing result in catastrophic deterioration of high voltage assemblies. Voids, air gaps, cracks, and separations provide a high probability for such deterioration and failure even with

optimum materials selection. Good packaging design and potting procedures minimizes the number and size of voids in insulation by optimizing accommodation for "easy" fill and gas escape.

Most manufacturers use vacuum impregnation followed by overpressure encapsulation, which greatly reduces void size and quantity. However, it is important that each material be evaluated for the correct vacuum and overpressure combination to obtain the least number of voids. For example, too long under vacuum can cause catalyst evaporation, thus degrading the resin within the insulation systems, resulting in accelerated dielectric failure. Good packaging design involves not only logical parts layout, but also the selection of a low enough viscosity potting compound to impregnate and bond the most difficult access elements without voids.

The following are recurrent failure items:

- a. Voids or gaps along insulation interfaces.
- b. Incomplete outgassing of air from the wiring and parts before applying the potting material.
- c. Gas release by the potting compound or encapsulated parts during cure caused voids within the potted volume.
- d. Cracks and voids between the outer winding and core of magnetic components results in partial discharges and arcing.
- e. Voids resulting from tapes and tabs used within magnetics.
- f. Air trapped by sleeving over component or between a wire and a sleeve.
- g. Cracks occurring from compounds with too high a glass transition temperature.
- h. Voids resulting from improper bonding of potting compound to components with incompatible surface preparation such as wax.

- i. Voids resulting from improper bonding of potting compound to structural surfaces, usually caused by contamination or insufficient preparation.
- j. Voids in printed wiring boards from mechanical stress in assembly processes.
- k. Debonding of laminated boards.
- l. Voids under screw points.
- m. Voids between an insulating board and a bolt or screw.

8.5 Materials and Processes. Materials and processes related problem areas are listed below.

MATERIAL PROBLEMS

- o High viscosity non-wettable resins
- o Poor compatibility of materials
- o Low thermal-conductivity resins
- o High thermal-expansion resins
- o High thermal stresses in resins
- o Poor thermal shock resins
- o Resins electrically operated at a dielectric polarization frequency
- o Short pot-life resins
- o Poorly adhering resins
- o Improper use of primers
- o Mold release contamination
- o High electrical loss resins
- o High shrinkage resins
- o Inadequate thermal stability of resins

PROCESS PROBLEMS

- o Mixing procedures
- o Application of primers
- o Vacuum-pressure cycles

- o Evacuation of resins
- o Workmanship
- o Cleaning and cleanliness control
- o Pre-drying of materials and parts
- o Equipment controls
- o Post-baking of encapsulated assemblies

MAGNETICS

- a. The encapsulant must completely wet and fill the internal fibers and voids between turns, and not crack or craze near the core or structural members.
- b. Tapes, sleeveings and wrappings persist as problems involving voids and material compatibility.
- c. Magnet wire intended for high reliability magnetics should be stored and handled with special protection from mechanical damage and contamination.
- d. By defining and imposing an adequate test program upon the producer of the required magnetic devices, a substantial increase in yield and a reduction in life cycle cost can be achieved. The following test categories are occasionally omitted or inadequately imposed:
 - 1. Electrical high-voltage tests
 - 2. Partial discharge and pulse tests after potting to determine insulation integrity between coils and between turns within a winding
 - 3. Production sampling to determine operation at ambient, altitude, and temperature extremes.

Connectors. Air gaps around pins and sockets and across insulating interfaces must be eliminated. Potting processes must insure that gas is not entrapped or surfaces contaminated.

Circuit Boards. Voids in circuit boards which can result in internal tracking and failure under high voltage conditions are often difficult to detect.

Resistors and Capacitors. Surfaces must bond to the encapsulant. Oil or wax-filled surfaces must be cleaned and prepared for proper bonding. Avoid hollow cores.

High Voltage Wire. Wire constructions with a semi-conducting layer over solid round wire conductors are preferred. Cables should have a semi-conducting layer next to the shield. This gives a more uniform field across the insulating system.

8.6 Overheating Of Components And Insulation.

High voltage assemblies use a relatively large amount of insulating material, and thermal management requirements increase somewhat exponentially, rather than proportionately. The design team must follow reliability derating criteria for the selection of components and materials for the temperatures involved, the following considerations are involved in minimizing life cycle costs and maximizing MTBF:

- a. Minimize thermal resistance from the module to ambient.
- b. Optimize the layout to minimize thermal paths within the module
- c. Utilize thermal spreaders where possible.
- d. Utilize high conductivity insulating material where possible.
- e. Utilize maximum allowable package size.
- f. Optimize layout to minimize thermal path from heat sources to heat sink.
- g. Utilize ceramic standoffs to provide improved thermal contact with the heat sink.
- h. Select materials which are thermally stable throughout the specified temperature range.

It is recommended that computer simulation be utilized to conduct a detailed thermal analysis for validating the packaging concept. When the design is declared feasible, and a prototype evaluated, then it is necessary to communicate with the manufacturing, test, and quality engineers to insure that thermal design integrity is maintained in the production process. Last, but vitally important, is the accelerated life test of an early production unit to learn of any latent degradations which might result from cumulative thermal effects at maximum electrical stress in conjunction with environmental testing.

A more comprehensive listing of problem areas and their solutions may be obtained in references 61 through 66.

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